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FINAL REPORT

PROPELLANT SELECTION FOR SPACECRAFT PROPULSION SYSTEMS

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VOLUME II
MISSIONS AND VEHICLES

PREPARED FOR
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
OFFICE OF ADVANCED RESEARCH AND TECHNOLOGY
WASHINGTON, D.C.

FOREWORD

This report was prepared by the Lockheed Missiles & Space Company, Sunnyvale, California, and contains the results of a study performed for the National Aeronautics and Space Administration, Office of Advanced Research and Technology, under Contract NASw-1644, Propellant Selection for Spacecraft Propulsion Systems. The report is printed in three volumes:

Volume I	Summary, Results, Conclusions, and Recommendations
Volume II	Missions and Vehicles
Volume III	Thermodynamics and Propulsion

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INTRODUCTION

This volume presents details of the mission analyses performed in Task I, leading to selection of two representative spacecraft stages, and of the design, structural, and performance analyses performed in Task II. Major sections in the volume include:

Section	Title
1	Mission Analysis – Task I
2	Stage Analysis Approach – Task II
3	Mars Orbiter Stage Investigation
4	Mars Excursion Module Ascent Stage Investigation
5	Operational Constraints
6	Sensitivity Analysis

Details of the two reference missions and spacecraft, as described in previous studies, are presented in the appendixes at the end of this volume.

Section 1
MISSION ANALYSIS – TASK I

The objective of the mission analysis phase was to define several propulsion vehicle systems for which space-storable propellants are competitive with other propellant combinations. This was accomplished by defining a broad spectrum of missions, applying a preliminary screening to these missions, and then conducting a simple analytical evaluation of each system in order to obtain the performance capability of each vehicle system/propellant combination. Figure 1 shows this procedure graphically.

All of the propulsive maneuvers for this broad spectrum of unmanned and manned interplanetary, lunar, and earth-orbit missions were defined. All of these cases were then assessed by the preliminary screening procedure to limit the number of cases for which a more detailed analysis and screening are required. The selected mission vehicle propulsion stages were then analyzed from an environmental consideration in order to evaluate and differentiate between the propellant combinations. Vehicle scaling laws and meteoroid criteria provided by NASA were used to conduct the analysis. Propulsion system data were collected and also requested from the supporting engine companies, Aerojet-General, Pratt & Whitney Aircraft, and Rocketdyne. The analytical procedures were mechanized by a simple computer program called RAPID, which defines all the vehicle parameters, in addition to initial weight, the parameter on which the comparison was made. Four vehicles were proposed as being promising for space-storable propellant applications from which the NASA Management Committee selected two for more detailed study. In addition, a preliminary sensitivity analysis and a propulsion commonality assessment were made during this phase.

1.1 MISSIONS

A representative group of potential missions was compiled, including both manned and unmanned systems for earth orbit, lunar, and interplanetary targets. A comprehensive

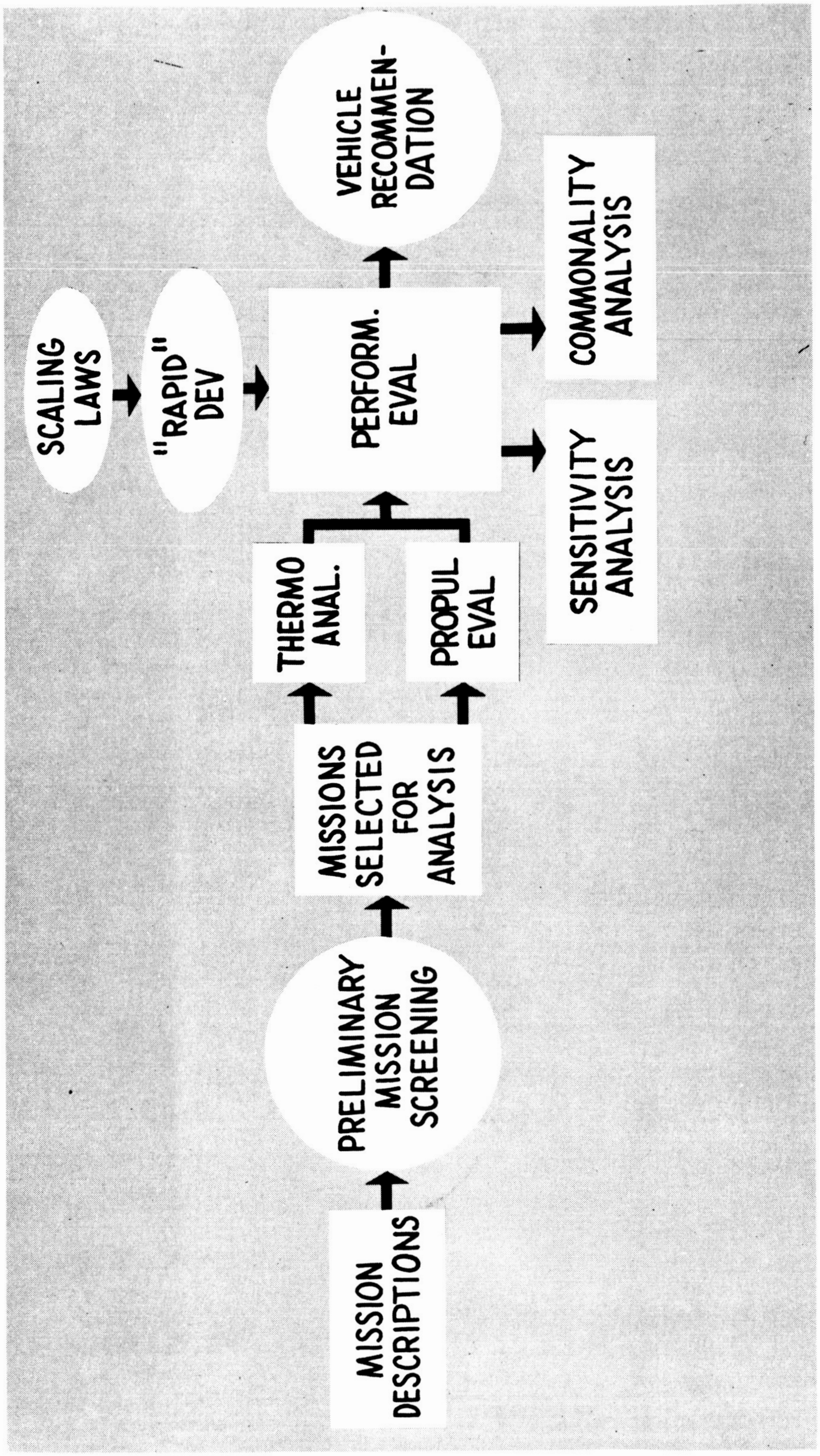


Fig. 1 Mission Analysis Sequence

set of documents was evaluated to obtain this information. The primary data sources are listed under the references, Section 7. The missions included in the basic mission list are identified and described in Appendix A. An identification code for missions, spacecraft stages, propulsion feed system, and propellants also is given in Appendix A.

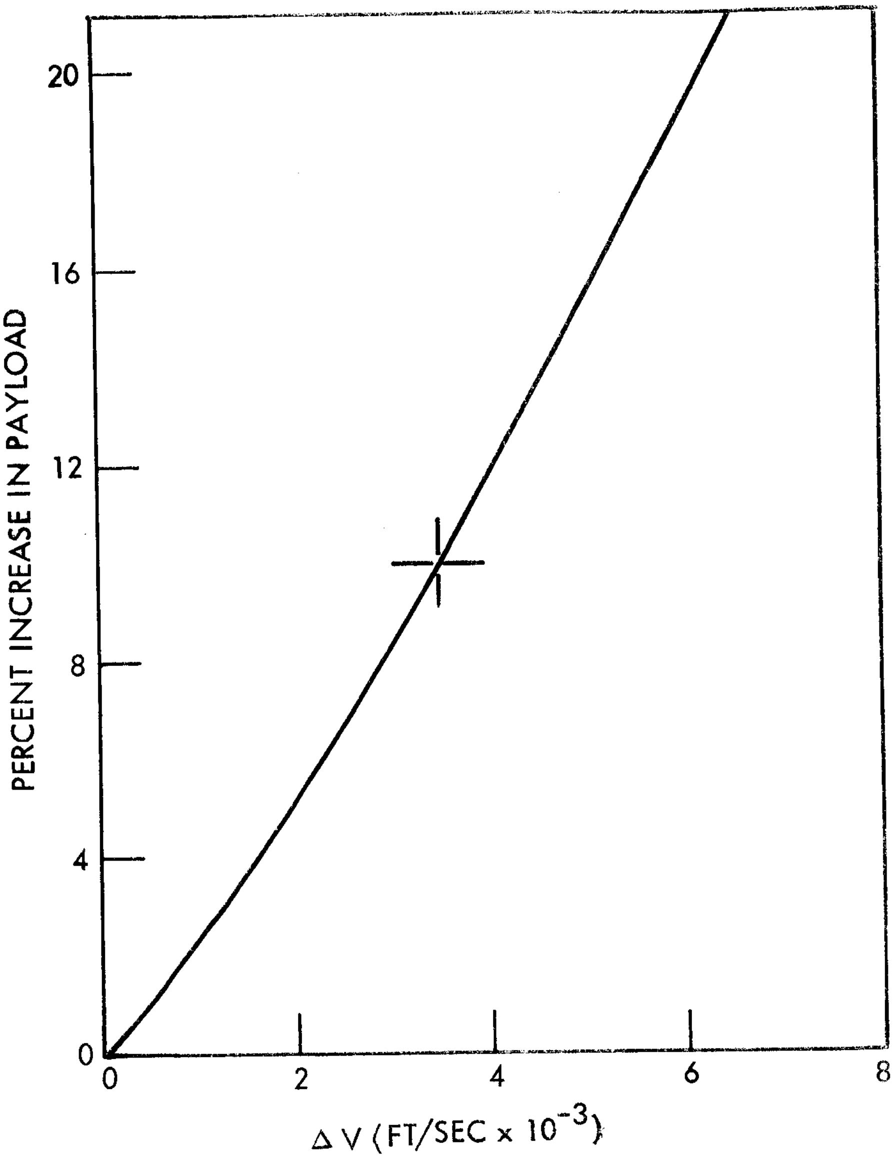
The missions described in Appendix A were then classified by propulsive maneuvers in order to conduct the preliminary screening.

To reduce the analysis of cases investigated to more manageable proportions, preliminary criteria were established for systems in which significant performance gains could be established. The criterion selected is that performance of the space-storable propulsion stage must be such that, for a fixed initial total system mass, the payload propelled by the space-storable stage is increased 10 percent or more over that obtained with the competing earth-storable or cryogenic system. Conversely, for a fixed payload mass the initial total system mass must be decreased 9 percent or more over that of the earth-storable or cryogenic system. This is an arbitrary criterion utilized only for initial screening of candidate stages in Task I.

To determine the applicability of this criterion, a comparison of earth-storable and space-storable systems was made on the basis of the total ΔV that the candidate propulsion stage must deliver. The following assumptions were made, and are generally favorable to retaining a space storable as a candidate propellant:

- Delivered I_{sp} of earth storables is 310 sec
- Delivered I_{sp} of space storables is 395 sec
- The propulsion system mass fraction for both earth storables and space storables is 0.85
- No propellant is lost through boiloff

The resulting percentage increase in payload mass for the space storable is plotted in Fig. 2 as a function of ΔV. From this figure it is seen that a 10-percent improvement in payload is not possible at ΔV's below about 3,500 ft/sec.



ASSUMPTIONS:

I_{sp} INCREASE FROM 310 TO 395 SEC

NO PROPELLANT BOILOFF

PROPULSION STAGE MASS FRACTION = 0.85

Fig. 2 Effect of Increase in Specific Impulse

The initial comparison of space storables with cryogenics is more complex and, generally, required thermal and structural analyses. Of the cryogenic stages associated with the specified reference missions, all are designed for high total ΔV capability. Thus, a screening on the basis of ΔV seemed inappropriate for this comparison, and each candidate was subjected to a preliminary systems definition through the use of scaling laws and standard models.

In applying these criteria the propulsive systems remaining to be analyzed are shown in Table 1. Without exception, the propulsive maneuvers are all the primary propulsive steps in each mission. This does not mean that the other propulsive maneuvers are not significant, but rather that systems designed for other purposes would be used for the secondary propulsive maneuvers.

1.2 PROPULSION DATA FOR TASK I

This analysis was conducted with propellants representative of cryogenics, space storables, and earth storables, rather than with all of the propellants. The propellants actually used in Task I and their liquidus range are shown in Fig. 3. The propulsion system characteristics utilized for this first analysis and used to classify the system characteristics are shown in Table 2.

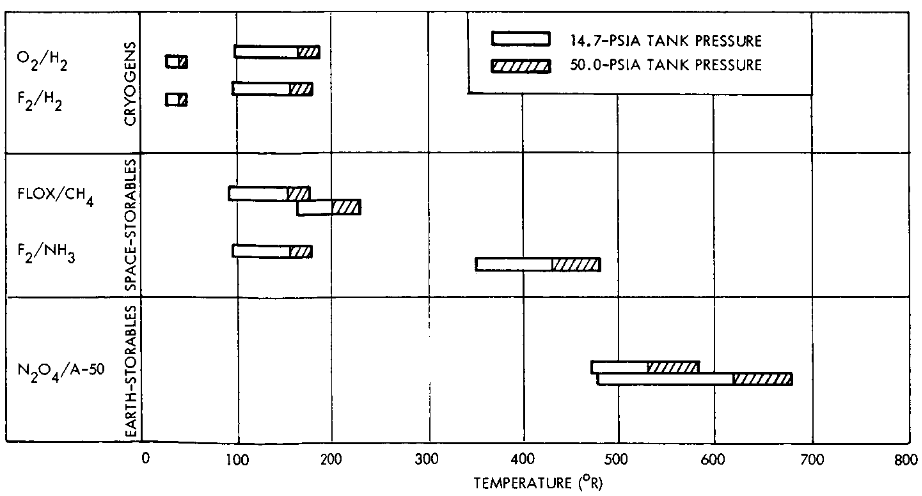


Fig. 3 Propellant Liquid Temperature Range

Table 1

SPACECRAFT STAGES SELECTED FOR ANALYSIS

Mission Name	Mission Code Number	Stage Code No.	Mission Year	Payload (lbm)	Stage Diameter (ft)	ΔV (ft/sec)	Exposure Time (days)	Thrust Nominal (lbf)	No. of Engine Starts
Voyager-Mars	MUO-1	OI	1973	8,143	21.6	6,950	195	8,000	6
Voyager-Mars, ABL	MUO-2	OI	1977	13,500	21.6	5,000	325	8,000	6
Voyager-Venus	VUO-1	OI	1977	4,500+ 2,500 Probe	10.0 to 21.6	13,500	140	8,000	6
Jupiter Orbiter	JUO-1	OI	1981	2,000	10.0	7,600	650	2,000	6
Saturn Orbiter	SUO-1	OI	1984	2,000	10.0	6,000	1,450	2,000	6
Lunar Manned Surface Station	LMS-1	PD	1978	19,340	21.6	9,186	178	15,000	3
Mars Manned Flyby	MMF-1	ED	1977	224,000	21.6	7,340	5	30,000	1
		Probe 1 (Orbiter)	1977	1,000	< 21.6	21,000	150	4,000	2
		Probe 3 (MSSR)	1977	< 100	< 21.6	36,000	150	Multi-stage	
Venus Manned Flyby	VMF-1	Probe 1 (Orbiter)	1977	1,500	< 21.6	13,000	115	4,000	2
Mars Manned Lander	MML-1	ED	1982	660,000	33.0	12,900	60 and 120	100K per Module	1
		AS	1982	80,000 (Gross)	< 33.0	15,500	221	50,000	4 min
Mars Manned Lander (Direct, 30-day stay)	MML-1	PD	1982	92,000	33.0	15,000	221	100K per Module	1
Mars Manned Lander (Swing-in, 30-day stay)	MML-2	ED	1982	770,000	33.0	12,700	60 and 120	100K per Module	1
		AS	1982	80,000 (Gross)	< 33.0	15,500	280	50,000	4
		PD	1982	92,000	33.0	16,000	280	100K per Module	1
Mars Manned Lander (Swing-in, 100-day stay)	MML-3	ED	1982	785,000	33.0	12,900	60 and 120	100K per Module	1
		AS	1982	80,000 (Gross)	< 33.0	15,500	300	50,000	4
		PD	1982	92,000	33.0	16,000	300	100K per Module	1
Venus Manned Orbiter	VMO-1	ED	1985	440,000	33.0	11,600	60 and 120	100K per Module	1
		PD	1985	92,000	33.0	14,000	173	100K per Module	1
Mars Manned Orbiter	MMO-1	ED	1980	634,000	33.0	13,700	60 and 120	100K per Module	1
		PD	1980	92,000	33.0	15,700	227	100K per Module	1
Earth Manned Orbiter	EMO-1	DS	1973	13,000	12.8	9,750	60	20,000	4

Table 2

PROPELLANTS ASSUMPTIONS FOR MISSION SCREENING

Parameter \ Propellant	F_2/H_2	O_2/H_2	$FLOX/CH_4$	F_2/NH_3	N_2O_4/A-50
Specific Impulse (sec)[a]	461	446	405	407	310
Mixture Ratio (O/F)	9:1	5:1	5.75:1	3.2:1	1.6:1

(a) I_{sp} was reduced to 95 percent of nominal for thrust levels below 8,000 lb.

1.3 THERMODYNAMIC ANALYSIS FOR TASK I

Preliminary predictions of propellant tank temperatures were generated for all missions analyzed in Task I. These temperatures were used to compute propellant boiloff in the performance evaluation. Passive thermal control techniques consisting of external thermal control surface finishes and insulation were employed to minimize heat transfer into the propellant and to maintain the propellants within their liquid ranges. Task I ground rules called for propellants initially at earth ambient pressure and saturated liquid condition with all subsequent heat input translated into propellant boiloff.

The temperatures computed represent the average temperature of the tank wall or the temperature of the outermost insulation surface in those cases where insulation is used. Heat transfer into the propellant is based upon the temperature difference across the insulation thickness. The internal temperature is assumed to be that of the liquid.

Average surface temperatures of the tanks were computed for Earth, Mars, and Venus orbits based on a random orientation, as shown in Fig. 4. Surface temperature for the interplanetary phases of applicable missions were computed considering that the

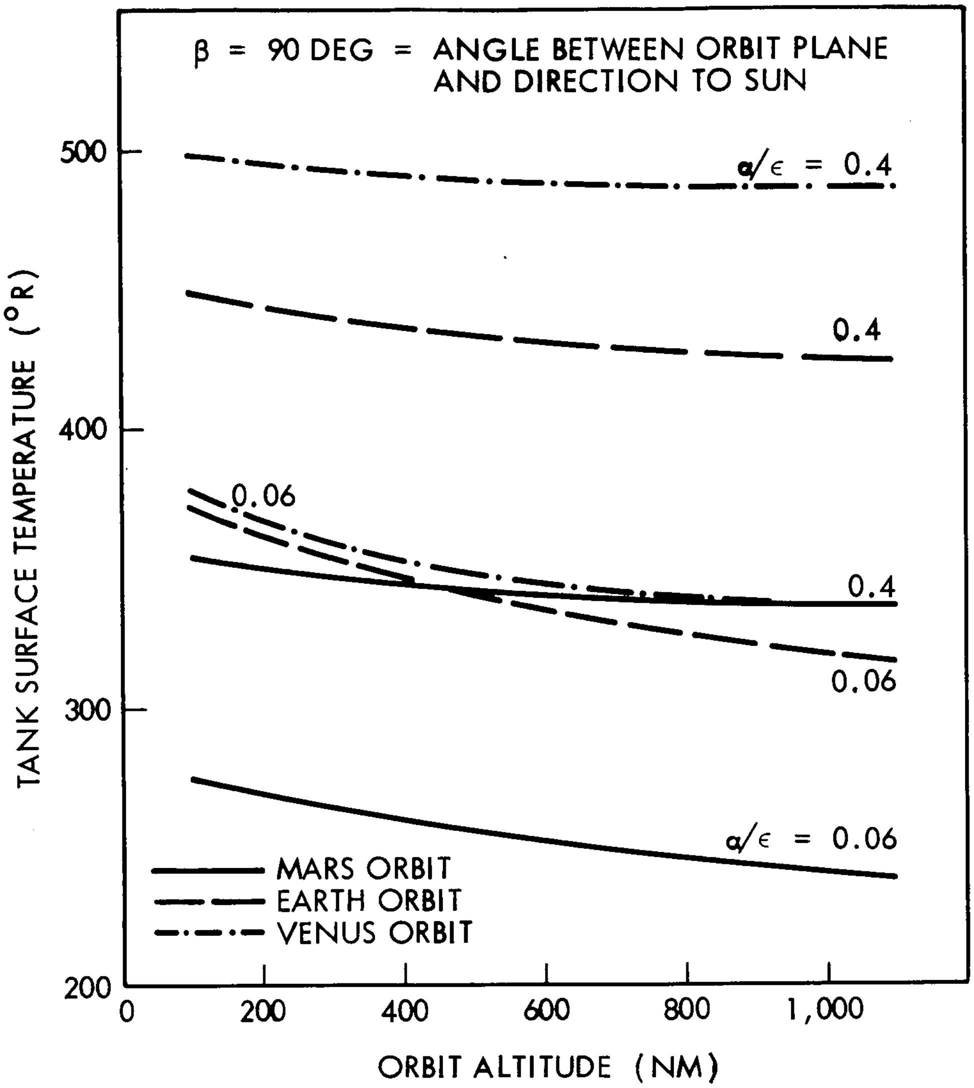


Fig. 4 Surface Temperature of Spherical Tank

propellant tanks were shaded from the sun (except earth-storables). Temperatures were varied according to the average distance the spacecraft was from the sun.

Two methods of external surface temperature control were considered: (1) variation of optical surfaces properties, i.e., α_s/ϵ ratio, and (2) orientation of the vehicle relative to the sun.

The energy balance between the tanks and their environment can be controlled by modulation of tank surface α_s and ϵ values. In cases where low temperatures were required, the tanks were assumed to be coated with an optical solar reflector (OSR $\alpha_s/\epsilon = 0.06$) or with a white paint that has undergone ultraviolet degradation ($\alpha_s/\epsilon = 0.4$). In an attempt to maintain the earth-storable propellants in their liquid temperature ranges, the α_s/ϵ ratios were increased, and it was assumed that modulation of the solar term could be accomplished by orientation of the vehicle.

Placing the vehicle in a nose to sun orientation (Fig. 5) during interplanetary coast so that the tanks are shaded by the spacecraft will eliminate the solar term from the tank energy balance, and reduced tank temperatures will result. In this orientation, the only parameters that will affect tank temperatures are the emittances of the tanks and forward bulkhead, the bulkhead temperature, and possibly energy from a nearby planet. Since the emittances of the OSR surface and the white paint are approximately the same, tanks with these coatings will achieve the same temperature levels when shaded.

In cases where an oxidizer and a fuel have different liquid ranges (e.g., F_2 and NH_3), the tanks must be thermally isolated. The fluorine tanks were designed to achieve low temperatures, and the ammonia tanks were assumed to be located where factors such as power dissipation and spacecraft skin temperature will maintain the fuel within its liquid temperature range.

Determination of tank surface temperatures during sun-oriented phases was based on a simplified thermal computer model consisting of a payload section (1 node at 540°R),

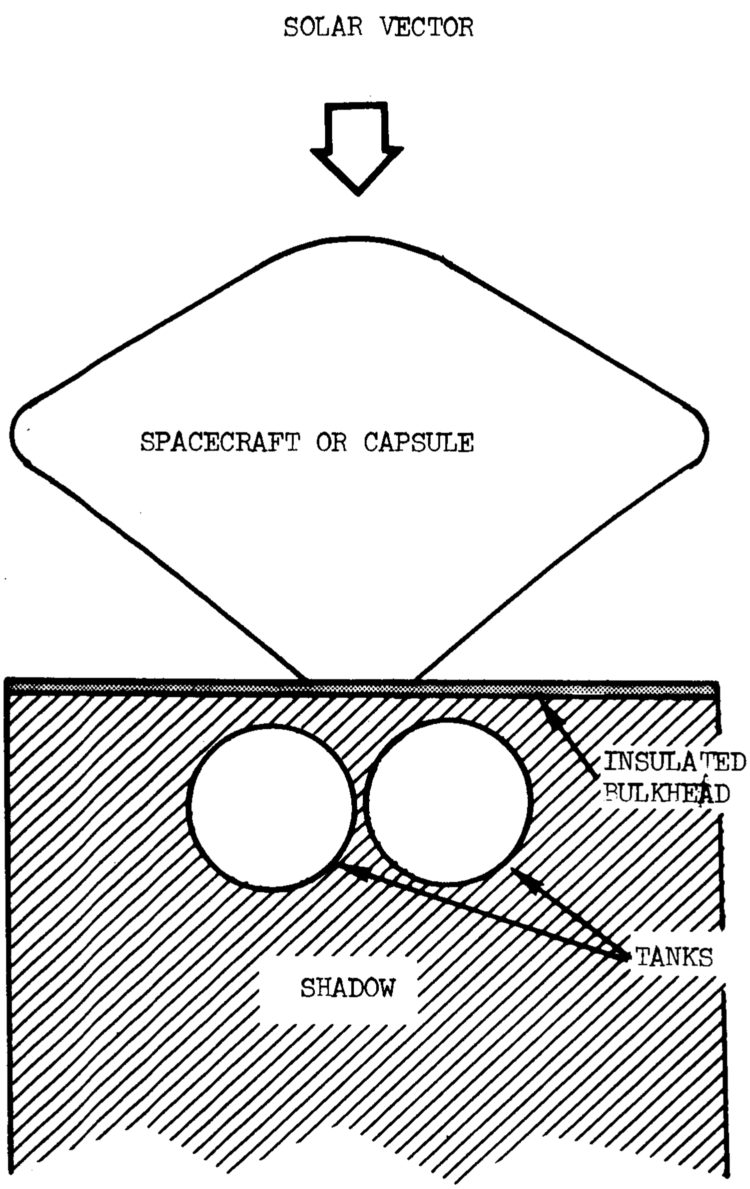


Fig. 5 Spacecraft Orientation During Cruise

an insulated bulkhead, and the propellant tanks. Radiation view factors between the bulkhead and tanks were computed. Conduction heat transfer from the payload and bulkhead to the propellant tanks was not considered in this phase.

Propellant tank temperatures for the various unmanned missions were found to be similar due to vehicle configurations and the assumptions made in this preliminary analysis. The temperatures of the propellant tanks in the shadow of the spacecraft were all determined to be approximately 160°R. In reality, there will be small differences in tank temperature due to tank-bulkhead geometry, which will influence the energy balance. For example, there will be less than a 5-deg Rankine temperature difference between two black ($\epsilon = 0.9$) tanks of 4-ft and 6-ft diameters if the bulkhead has a low emittance ($\epsilon = 0.05$) surface. Passive thermal control techniques can be optimized to further reduce tank energy levels. The use of 50-percent aluminum ($\epsilon = 0.05$) and 50-percent black pattern ($\epsilon = 0.9$) on the tanks in conjunction with a low emittance coating on the bulkhead may yield lower tank temperatures in the range of 110° to 130°R for the probe missions.

Propellant temperature levels for the manned missions tend to be higher than for the probe missions because the diameters of the tanks are larger relative to the bulkhead diameter.

Propellant tank surface temperatures during a lunar stay are presented in Fig. 6 for α_s/ϵ ratios of 0.4 and 0.06. Since the lunar temperature is below the boiling points of CH_4, NH_3, and N_2O_4 during certain portions of the lunar period, the time-averaged surface temperatures were computed to determine net heat input to the propellant.

Preliminary tank surface temperatures for the five propellant combinations investigated, and the assumptions made to generate these temperatures, are presented in Table 3 for the missions under consideration. For the manned mission planetary probes, a nominal "cold case" and a more conservative "hot case" were considered.

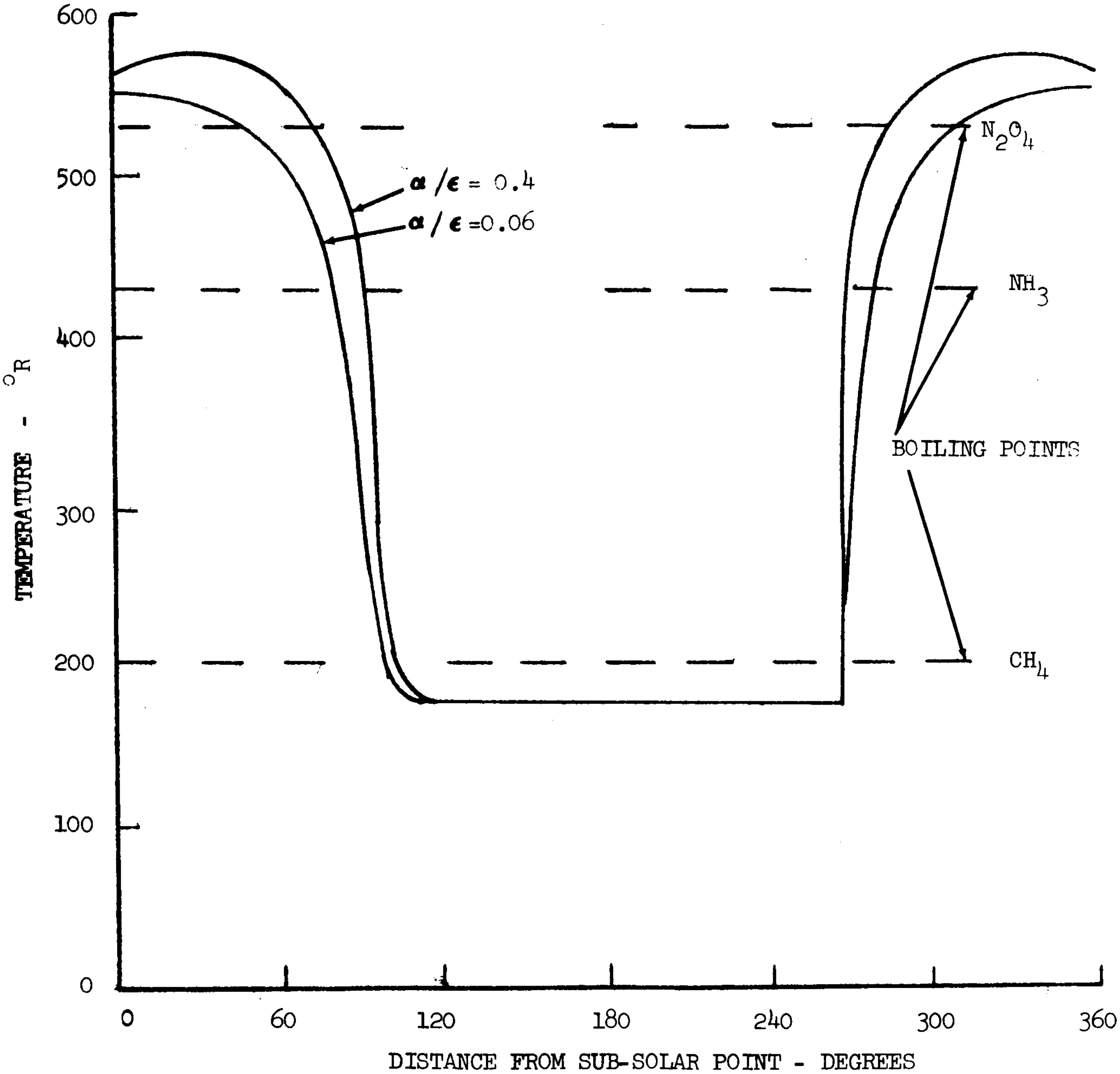


Fig. 6 Tank Temperature Vs. Position on Lunar Surface

Table 3

TANK SURFACE TEMPERATURES – TASK I ANALYSIS

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Voyager – Mars-1 (195-Day Transit)				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.5 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
Voyager – Mars-2 (325-Day Transit)				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.5 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
Voyager – Venus-1 (140-Day Transit)				
N_2O_4/A-50	500	α/ϵ = 0.59 with orientation	500	α/ϵ = 0.59 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.23 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Jupiter Orbiter (650-Day Transit)				
N_2O_4/A-50	<500	high α/ϵ with orientation	<500	high α/ϵ with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	high α/ϵ with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	$\alpha/=$ = 0.06, or 0.4 with shading
Saturn Orbiter (1450-Day Transit)				
N_2O_4/A-50	<500	high α/ϵ with orientation	<500	high α/ϵ with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	high α/ϵ with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
Mars Manned Flyby (150-Day Transit) "Cold Case"				
N_2O_4 A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	200	α/ϵ = 0.06, or 0.4 with shading	200	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	200	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.5 with location
O_2-H_2	200	α/ϵ = 0.06, or 0.4 with shading	200	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	200	α/ϵ = 0.06, or 0.4 with shading	200	α/ϵ = 0.06, or 0.4 with shading

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Mars Manned Flyby (150-Day Transit) "Hot Case"				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	320	α/ϵ = 0.4 enclosed	320	α/ϵ = 0.4 enclosed
F_2-NH_3	320	α/ϵ = 0.4 enclosed	400	α/ϵ = 0.5 with location
O_2-H_2	320	α/ϵ = 0.4 enclosed	320	α/ϵ = 0.4 enclosed
F_2-H_2	320	α/ϵ = 0.4 enclosed	320	α/ϵ = 0.4 enclosed
Venus Manned Flyby (115-Day Transit) "Cold Case"				
N_2O_4/A-50	500	α/ϵ = 0.59 with orientation	500	α/ϵ = 0.59 with orientation
FLOX-CH_4	200	α/ϵ = 0.06, or 0.4 with shading	200	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	200	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.23 with location
O_2-H_2	200	α/ϵ = 0.06, or 0.4 with shading	200	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	200	α/ϵ = 0.06, or 0.4 with shading	200	α/ϵ = 0.06, or 0.4 with shading
Venus Manned Flyby (115-Day Transit) "Hot Case"				
N_2O_4/A-50	500	α/ϵ = 0.59 with orientation	500	α/ϵ = 0.59 with orientation
FLOX-CH_4	430	α/ϵ = 0.4 enclosed	430	α/ϵ = 0.4 enclosed
F_2-NH_3	430	α/ϵ = 0.4 enclosed	400	α/ϵ = 0.23 with location
O_2-H_2	430	α/ϵ = 0.4 enclosed	430	α/ϵ = 0.4 enclosed
F_2-H_2	430	α/ϵ = 0.4 enclosed	430	α/ϵ = 0.4 enclosed

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Venus Manned Orbiter Planet Departure (30-Day E.O.)*				
N_2O_4/A-50	450	$\alpha/\epsilon = 0.4$	450	$\alpha/\epsilon = 0.4$
FLOX-CH_4	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
F_2-NH_3	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	400	$\alpha/\epsilon = 0.25$ with location
O_2-H_2	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
F_2-H_2	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
Venus Manned Orbiter Planet Departure (113-Day Transit)				
N_2O_4/A-50	<500	$\alpha/\epsilon = 0.4$ with orientation	<500	$\alpha/\epsilon = 0.4$ with orientation
FLOX-CH_4	160	$\alpha/\epsilon = 0.06$, or 0.4 with shading	160	$\alpha/\epsilon = 0.06$ or 0.4 with shading
F_2-NH_3	160	$\alpha/\epsilon = 0.06$, or 0.4 with shading	400	$\alpha/\epsilon = 0.25$ with location
O_2-H_2	160	$\alpha/\epsilon = 0.06$, or 0.4 with shading	160	$\alpha/\epsilon = 0.06$, or 0.4 with shading
F_2-H_2	160	$\alpha/\epsilon = 0.06$, or 0.4 with shading	160	$\alpha/\epsilon = 0.06$, or 0.4 with shading

*E.O. = Earth Orbit Phase

Table 3 (Cont.)

Mission Propellants	Oxidizer		Fuel	
	Temperature (°R)	Assumptions	Temperature (°R)	Assumptions
Venus Manned Orbiter Planet Departure (30-Day V. O.)*				
N_2O_4/A-50	500	$\alpha/\epsilon = 0.4$	500	$\alpha/\epsilon = 0.4$
FLOX-CH_4	350 500	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 500	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
F_2-NH_3	350 500	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	400	$\alpha/\epsilon = 0.25$ with location
O_2-H_2	350 500	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 500	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
F_2-H_2	350 500	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 500	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
Mars Manned Lander-3 Planet Departure (30-Day E. O.)**				
N_2O_4/A-50	500	$\alpha/\epsilon = 1.25$ with orientation	500	$\alpha/\epsilon = 1.25$ with orientation
FLOX-CH_4	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
F_2-NH_3	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	400	$\alpha/\epsilon = 0.5$ location
O_2-H_2	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$
F_2-H_2	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$	350 450	$\alpha/\epsilon = 0.06$ $\alpha/\epsilon = 0.4$

*V. O. = Venus Orbit Phase
**E. O. = Earth Orbit Phase

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Mars Manned Lander-3 Planet Departure (170-Day Transit)				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.5 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
Mars Manned Lander-3 Planet Departure (100-Day M. O)*				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	250 350	α/ϵ = 0.06 α/ϵ = 0.4	250 350	α/ϵ = 0.06 α/ϵ = 0.4
F_2-NH_3	250 350	α/ϵ = 0.06 α/ϵ = 0.4	400	α/ϵ = 0.5 with location
O_2-H_2	250 350	α/ϵ = 0.06 α/ϵ = 0.4	250 350	α/ϵ = 0.06 α/ϵ = 0.4
F_2-H_2	250 350	α/ϵ = 0.06 α/ϵ = 0.4	250 350	α/ϵ = 0.06 α/ϵ = 0.4

*M.O. = Mars Orbit Phase

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Mars Manned Lander-1 Ascent Stage (30-Day E.O.)*				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	350 450	α/ϵ = 0.06 α/ϵ = 0.4	350 450	α/ϵ = 0.06 α/ϵ = 0.4
F_2-NH_3	350 450	α/ϵ = 0.06 α/ϵ = 0.4	400	α/ϵ = 0.5 with location
O_2-H_2	350 450	α/ϵ = 0.06 α/ϵ = 0.4	350 450	α/ϵ = 0.06 α/ϵ = 0.4
F_2-H_2	350 450	α/ϵ = 0.06 α/ϵ = 0.4	350 450	α/ϵ = 0.06 α/ϵ = 0.4
Mars Manned Lander-1 Ascent Stage (161-Day Transit)				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.5 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading

*E.O. = Earth Orbit Phase

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Mars Manned Lander-1 Ascent Stage (30-Day Mars Surface)				
N_2O_4/A-50	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.0 or 0.4
FLOX-CH_4	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
F_2-NH_3	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
O_2-H_2	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
F_2-H_2	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
Mars Manned Lander-3 Earth Departure (60- and 120-Day E.O.)				
N_2O_4/A-50	500	α/ϵ = 0.5 with orientation	500	α/ϵ = 0.5 with orientation
FLOX-CH_4	350 450	α/ϵ = 0.06 α/ϵ = 0.4	350 450	α/ϵ = 0.06 α/ϵ = 0.4
F_2-NH_3	350 450	α/ϵ = 0.06 α/ϵ = 0.4	400	α/ϵ = 0.3 with location
O_2-H_2	350 450	α/ϵ = 0.06 α/ϵ = 0.4	350 450	α/ϵ = 0.06 α/ϵ = 0.4
F_2-H_2	350 450	α/ϵ = 0.06 α/ϵ = 0.4	350 450	α/ϵ = 0.06 α/ϵ = 0.4

*E.O. = Earth Orbit Phase

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Mars Manned Lander-1 Planet Departure (30-Day E. O.)*				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	350	α/ϵ = 0.06	350	α/ϵ = 0.06
	450	α/ϵ 0.4	450	α/ϵ = 0.4
F_2-NH_3	350	α/ϵ = 0.06	400	α/ϵ = 0.5 with location
	450	α/ϵ = 0.4		
O_2-H_2	350	α/ϵ = 0.06	350	α/ϵ = 0.06
	450	α/ϵ = 0.4	450	α/ϵ = 0.4
F_2-H_2	350	α/ϵ = 0.06	350	α/ϵ = 0.06
	450	α/ϵ = 0.4	450	α/ϵ = 0.4
Mars Manned Lander-1 Planet Departure (161-Day Transit)				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.5 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading

*E. O. = Earth Orbit Phase

Table 3 (Cont.)

Mission / Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Mars Manned Lander-1 Planet Departure (30-Day M. O.)*				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	250	α/ϵ = 0.06	250	α/ϵ = 0.06
	350	α/ϵ = 0.4	350	α/ϵ = 0.4
F_2-NH_3	250	α/ϵ = 0.06	400	α/ϵ = 0.5 with location
	350	α/ϵ = 0.4		
O_2-H_2	250	α/ϵ = 0.06	250	α/ϵ = 0.06
	350	α/ϵ = 0.4	350	α/ϵ = 0.4
F_2-H_2	250	α/ϵ = 0.06	250	α/ϵ = 0.06
	350	α/ϵ = 0.4	350	α/ϵ = 0.4
Mars Manned Lander-3 Ascent Stage (30-Day E. O.)**				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	350	α/ϵ = 0.06	350	α/ϵ = 0.06
	450	α/ϵ = 0.4	450	α/ϵ = 0.4
F_2-NH_3	350	α/ϵ = 0.06	400	α/ϵ = 0.5 with location
	450	α/ϵ = 0.4		
O_2-H_2	350	α/ϵ = 0.06	350	α/ϵ = 0.06
	450	α/ϵ = 0.4	450	α/ϵ = 0.4
F_2-H_2	350	α/ϵ = 0.06	350	α/ϵ = 0.06
	450	α/ϵ = 0.4	450	α/ϵ = 0.4

*M.O. = Mars Orbit Phase
**E.O. = Earth-Orbit Phase

Table 3 (Cont.)

Mission/ Propellants	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Mars Manned Lander-3 Ascent Star (170-Day Transit)				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	400	α/ϵ = 0.05 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
Mars Manned Lander-3 Ascent Star (100-Day Mars Surface)				
N_2O_4/A-50	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
FLOX-CH_4	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
F_2-NH_3	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
O_2-H_2	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
F_2-H_2	530	α/ϵ = 0.06 or 0.4	530	α/ϵ = 0.06 or 0.4
Earth Manned Orbiter-1 Descent Stage (60-Day E.O.)*				
N_2O_4/A-50	500	α/ϵ = 1.25 with orientation	500	α/ϵ = 1.25 with orientation
FLOX-CH_4	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-NH_3	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.5 with location
O_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading
F_2-H_2	160	α/ϵ = 0.06, or 0.4 with shading	160	α/ϵ = 0.06, or 0.4 with shading

*E.O. = Earth Orbit Phase

Table 3 (Cont.)

Mission/	Oxidizer Temperature (°R)	Oxidizer Assumptions	Fuel Temperature (°R)	Fuel Assumptions
Lunar Manned Station (θ = 177 Days)* α/ϵ = 0.06)				
N_2O_4/A-50	545 280	For 43.4 Days For 133.6 Days	310	For 177 Days
FLOX CH_4	310	For 177 Days	475 185	For 96 Days For 81 Days
F_2-NH_3	310	For 177 Days	520 190	For 87 Days For 90 Days
O_2-H_2	310	For 177 Days	310	For 177 Days
F_2-H_2	310	For 177 Days	310	For 177 Days
Lunar Manned Station (θ = 177 Days)* (α/ϵ = 0.4)				
N_2O_4/A-50	560 235	For 73.9 Days For 103.1 Days	330	For 177 Days
FLOX-CH_4	330	For 177 Days	510 190	For 98.4 Days For 78.6 Days
F_2-NH_3	330	For 177 Days	550 195	For 91.7 Days For 85.3 Days
O_2-H_2	330	For 177 Days	330	For 177 Days
F_2-H_2	330	For 177 Days	330	For 177 Days

θ = Lunar Surface Stay Time

Tank surface temperatures computed during this preliminary phase of the study were considered adequate for general propellant performance comparisons. More detailed analyses were conducted in Task II.

Propellant boiloff during interplanetary-coast phases probably can be eliminated for all oxidizers and fuels considered, except for hydrogen, in those designs where tanks are exposed to free space and shaded by the payload. Some propellants require special thermal control (i. e., insulation) to prevent freezing.

1.4 PERFORMANCE EVALUATION

The performance evaluation of all the combinations of vehicles and propellants was conducted utilizing basic scaling-law relationships. The mission parameter, engine data, and thermodynamic analyses served as input data for these calculations. The various cases are compared on performance with system initial weight as the parameter. The study ground rules for Task I required that all net propellant heating be translated into boiloff. The relationships utilized in the analysis are as follows:

The initial weight was computed for each case using the rocket equation

$$\Delta V = I_{sp} \, g \ell n \frac{W_o}{W_F} \tag{1}$$

The initial weight is

$$W_O = W_{PL} + W_P + W_S + W_E + W_I + W_{BO} + W_M \tag{2}$$

where

ΔV = velocity increment for the propulsive step (ft/sec)
I_{sp} = specific impulse of propellant combination (lbf/sec/lbm)
W_O = initial weight (lb)

W_F = final weight (lb)
W_{PL} = payload weight (lb)
W_P = propellant weight (lb)
W_S = propulsion system weight (lb)
W_E = engine weight (lb)
W_I = insulation weight (lb)
W_{BO} = boiloff weight (lb)
W_M = meteoroid shield weight (lb)
g = 32.2 ft/sec^2

The system weight is defined as

$$W_S = \frac{0.1\, W_P^{0.9}}{\sigma_M^{0.533}} + 1,100 \text{ lb} \tag{3}$$

where σ_M = specific gravity of the propellant mixture.

The engine weight for the engine sizes considered is defined as

$$W_E = 0.0125T + 100 \text{ lb} \tag{4}$$

where T = engine thrust in lb.

The insulation and boiloff weight are optimized and can be expressed as

$$W_I = \rho_I\, A\, \delta_I$$

where

$$\delta_{I_i} = \sqrt{\frac{K(1 + C)}{h_i \rho_I}\left\{\left[\sum_{\text{Stage 1}}^{i}\left(\frac{1 + \prod_{\text{Stage 1}}^{i-1} \mu_i \left(\frac{d_{W_s}}{d_{W_p}}\right)_i}{\prod_{\text{Stage 1}}^{i} \mu_i}\right)\right.\right.} \\ \left.\left.\sum_{\text{Phase 1}}^{j}\left(T_{S_j} - T_{BP_i}\right)(t_j - t_{j-1})\right]\right\}^{1/2} \tag{5}$$

$$W_{BO_i} = \frac{K(1 + C)\,A_i}{h_i\,\rho_{I_i}}\left\{\sum_{\text{Phase 1}}^{i}\left(T_{s_j} - T_{BP_i}\right)(t_j - t_{j-1})\right\} \tag{6}$$

where

ρ_I = density of insulation (4 lb/ft^3)

A = tank surface area (ft^2)

δ_I = thickness of insulation (ft)

K = conductivity of insulation (3×10^{-5} Btu/hr-ft-°R)

C = tank conduction parameter (0. 2)

h = propellant heat of vaporization (Btu/lb)

μ = mass ratio

$$\left(\frac{d_{W_s}}{d_{W_p}}\right)_i = \frac{0.09}{\left(W_P + W_{BO}\right)_i^{0.1}}\,\sigma_M^{0.533}$$

T_S = tank surface temperature (°R)

T_{BP} = temperature of propellant boiling point (°R)

t = time (days)

The following subscripts are used:

i = stage number

j = mission phase

The meteoroid shield is defined as

$$W_M = 1.195 \times 10^{-2} \, C \, (At_x)^{0.352} \, A$$

where

$$C = \begin{array}{l} 1 < r\sigma \\ 2 > r\sigma \end{array}$$

t_x = exposure time (days)

The numerical values for propellant change of state and heats of vaporization used in the computation are given in Table 4.

In all cases during this task, all the heat that entered into a propellant tank was assumed to result in boiloff.

The performance calculations were made with the aid of a small computer program entitled RAPID (Rapid Analysis of Propellants for Initial Design). This program mechanized performance computations by using closed form expressions for weight expendables rather than using an integrating technique. A summary of the initial weights for all the cases investigated is given in Table 5.

Table 4

PROPELLANT TEMPERATURES AND HEATS OF VAPORIZATION

Fuel or Oxidizer	Temperature		Heat of Vaporization (Btu/lb)
	Boiling Point (°R)	Freezing Point (°R)	
N_2O_4	529.7	471.4	178.2
A-50	619.6	478.4	425.8
FLOX (82.3% F_2, 17.7% O_2)	154.0	96.6	75.0
CH_4	200.9	163.1	219.4
F_2	153.0	96.3	71.5
NH_3	431.6	351.7	596.2
O_2	162.2	98.5	91.6
H_2	36.4	25.1	195.3

Table 5

STAGE INITIAL WEIGHT BY STAGE AND PROPELLANT

Stage[a]	Initial Weight (lb)				
	F_2/H_2	O_2/H_2	N_2O_4/A-50	FLOX/CH_4	F_2/NH_3
MUO-1-OI	8,150	8,800	11,720	8,600	8,520
MUO-2-OI	8,500	9,150	11,700	8,830	8,760
VUO-1-OI	12,410	13,920	21,010	13,430	13,240
JUO-1-OI	4,750	5,350	5,840	4,430	4,380
SUO-I-OI	4,280	4,840	4,530	3,620	3,580
LMS-1-PD	23,630	26,190	36,160	25,610	25,070
MMF-1 OI 1	13,380	15,700	26,850	14,690	14,400
MMF-1 OI 2	5,810	6,410	8,860	6,200	6,120
MML-1-ED	183,030	200,940	334,030	212,350	208,930
MML-1-AS	39,870	45,960	70,810	43,710	42,620
MML-1-PD	201,380	225,340	383,330	232,770	228,500
VMO-1-PD	178,920	198,770	331,930	205,350	201,700
VMF-1-OI	6,930	7,830	10,970	7,310	7,200
EMO-1-DS	17,280	18,990	28,050	19,110	18,900

[a]Stage codes are defined in Appendix A.

The initial weights were then normalized using N_2O_4/A-50 as a reference. The results are presented in Table 6. Values greater than 100 indicate a heavier (poorer performance) system then N_2O_4/A-50, and less than 100 a lighter (better performance) system.

The results of the analysis indicated the following:

- Space storables outperformed earth storables by 25 to 74 percent
- Performance of space storables and O_2/H_2 was within ±10 percent of each other for most missions
- F_2/H_2 outperformed the space storables by 4 to 17 percent except for the Jupiter and Saturn missions where FLOX/CH_4 performance was better than F_2/H_2 by 7 and 18 percent, respectively.

1.5 SENSITIVITY ANALYSIS FOR TASK I

The basic performance analysis was conducted with nominal mission parameters. It was of interest, therefore, to assess the effects of changes in assumptions concerning spacecraft structures and thermodynamics. Table 7 shows the effects of these parameters. The first column shows the basic mission performance ranking. The column marked "hot" utilized the higher α_s/ϵ values and temperatures shown in Table 4, the column marked "long" assumed a 120-day earth-orbit residence rather than the normal 60-day residence, and the column marked "heavy" used structural factors that were double the nominal values. There was very little effect of the perturbations imposed by the sensitivity analysis. On some of the manned missions using the heavy structural factors, the space storables displaced O_2/H_2 from second place.

1.6 COMMONALITY ANALYSIS FOR TASK I

A commonality analysis was conducted in which a comparison of thrust levels and propellant loadings for stages in which space storables appeared competitive. These systems are grouped in Table 8 into four distinct categories, with specific applications for each classification identified.

Table 6

PROPELLANT PERFORMANCE BY MISSION

MISSION	STAGE WT (% OF N_2O_4/A-50)[a]			
	F_2/H_2	O_2/H_2	FLOX/CH_4	F_2/NH_3
SATURN UNMANNED ORBITER	95	107	80	79
JUPITER UNMANNED ORBITER	81	92	76	75
VENUS MANNED FLYBY – ORBITER PROBE	63	71	67	66
MARS MANNED FLYBY – ORBITER PROBE	50	58	55	54
MARS MANNED LANDER – MEM ASCENT STAGE	56	65	62	60
VENUS UNMANNED ORBITER	59	66	64	63
MARS UNMANNED ORBITER	70	75	73	73
LUNAR MANNED SURFACE STATION – RETURN STAGE	66	73	71	69
EARTH MANNED SYNCHRONOUS ORBITER – DESCENT STAGE	62	68	68	67
MARS MANNED LANDER – PLANET DEPARTURE STAGE	53	59	61	60
VENUS MANNED ORBITER – PLANET DEPARTURE STAGE	54	60	62	61
MARS MANNED LANDER – EARTH DEPARTURE STAGE	55	60	64	63

(a) Propulsion stage weight compared with N_2O_4/A-50 stage.

Table 7

PERFORMANCE SENSITIVITY ASSESSMENT

MISSION	BASIC	HOT	LONG	HEAVY
MUO-1	1, 0, 8, 2, 3	---	---	1, 0, 8, 2, 3
MUO-2	1, 0, 8, 2, 3	---	---	1, 0, 8, 2, 3
VUO-1	1, 0, 8, 2, 3	---	---	1, 0, 8, 2, 3
JUO-1	0, 8, 1, 2, 3	---	---	0, 8, 1, 2, 3
SUO-1	0, 8, 1, 3, 2	---	---	0, 8, 1, 3, 2
LMS-1	1, 0, 8, 2, 3	1, 0, 8, 2, 3	---	1, 0, 8, 2, 3
MMF-1 P1	1, 0, 8, 2, 3	1, 0, 8, 2, 3	---	
VMF-1 P1	1, 0, 8, 2, 3	0, 1, 8, 2, 3	---	0, 1, 8, 2, 3
MML-1 ED	1, 2, 0, 8, 3	1, 2, 0, 8, 3	1, 2, 0, 8, 3	1, 0, 8, 2, 3
AS	1, 0, 8, 2, 3	1, 0, 8, 2, 3	1, 0, 8, 2, 3	1, 0, 8, 2, 3
PD	1, 2, 0, 8, 3	1, 2, 0, 8, 3	1, 2, 0, 8, 3	1, 0, 8, 2, 3
VMO-1 PD	1, 2, 0, 8, 3	1, 2, 0, 8, 3	---	1, 0, 8, 2, 3
EMO-1 DS	1, 0, 2, 8, 3	---	---	1, 0, 8, 2, 3

PROPELLANTS:
1 – F_2/H_2
2 – O_2/H_2
3 – N_2O_4/A-50
8 – FLOX/CH_4
0 – F_2/NH_3

Table 8

PROPULSION SYSTEM COMMONALITY

THRUST RANGE (LB)	PROPELLANT LOADING – $FLOX/CH_4$ (LB)	STAGES
2,000 TO 4,000	2,000 TO 5,000	(1) ORBIT INJECTION AT JUPITER AND SATURN (2) SECOND STAGE OF PROBE ORBIT INJECTION AT MARS FROM MANNED FLYBY
8,000	7,000 TO 13,000	(1) ORBIT INJECTION AT MARS AND VENUS (2) FIRST STAGE OF PROBE ORBIT INJECTION AT MARS FROM MANNED FLYBY
15,000 TO 20,000	10,000 TO 22,000	(1) MANNED DEPARTURE FROM LUNAR SURFACE (2) SECOND STAGE OF MEM ASCENT (3) MANNED DESCENT FROM SYNCHRONOUS EARTH ORBIT
30,000 TO 50,000	15,000 TO 24,000	(1) FIRST STAGE OF MEM ASCENT

1.7 VEHICLES RECOMMENDED FOR USE IN TASK II ANALYSIS

Subsequent to the performance and sensitivity computations, several stages were recommended for further analysis on the basis that they had an attractive potential for space-storable application. These stages are listed in Table 9. The NASA Management Committee selected the Unmanned Mars Orbiter – Orbit Injection Stage and the Mars Excursion Module (MEM) – Ascent Stage as being the most qualified for further study. This selection was based, in part, on vehicle size, application, and available documentation. There was also a slight variation in the propellants selected for application to these two stages. The propellants and engine feed systems selected for the various cases are shown in Table 10.

Table 9

STAGES RECOMMENDED AS TASK II CANDIDATES

Stage	Thrust (lb)	Payload (lb)	ΔV (ft/sec)	Duration (Days)
Unmanned Saturn Orbiter – Orbit Injection	2,000	2,000	6,000	1,450
Unmanned Mars Orbiter – Orbit Injection	8,000	8,143	6,950	195
Lunar Manned Station – Return Stage	15,000	19,340	9,186	178
Mars Excursion Module – Ascent Stage*	50,000	13,500	15,500	220
*(Revised for Task II)	(30,000)	(5,260)	(16,000	(221)

Table 10

PROPELLANT AND ENGINE FEED COMBINATIONS FOR TASK II

Propellants	MARS ORBITER		MEM ASCENT
	Pump Fed	Pressure Fed	Pump Fed
F_2/H_2	X	X	X
O_2/H_2	X	X	X
$FLOX/CH_4$	X	X	X
OF_2/CH_4	X	X	X
OF_2/B_2H_6	–	X	*
F_2/NH_3	X	X	X
N_2O_4/A-50	X	X	–
ClF_5/MHF-5	X	X	X

*Preliminary assessment only, using pressure feed.

Section 2
STAGE ANALYSIS APPROACH – TASK II

The analysis of alternate propellants for a spectrum of space missions requires a baseline vehicle that can be adapted for the various propellants according to a standardized procedure. This was accomplished by gathering data and reviewing the reference configurations for the two stages chosen by NASA for detailed investigation in task II. The baseline vehicles are described in Appendixes A, B, and C. The propulsion stages were then modified to use each of the candidate propellants in turn.

The modified designs were subjected to a detailed thermal, pressurization, structural, and propulsion system analysis using proven techniques for optimizing tank design, support strut design, insulation thickness, and pressurization system design. A performance analysis was then conducted, and the final designs were compared with the reference designs in order to highlight the merits of each propellant as compared to others. The step-by-step procedure followed in the analysis is presented in Fig. 7.

2.1 PROCEDURE FOR STAGE ANALYSIS – TASK II

2.1.1 Conceptual Design

The first step was to assess all the design concepts that have been developed for the selected missions. The level of definition available had a strong influence on the effort required for the next step. The Mars Orbiter as studied by TRW had been developed in great detail, while the selected Mars Excursion Module configuration had not been studied in similar depth.

The baseline vehicle plus the design criteria to be used for the vehicle adaptation was then incorporated into the conceptual design. The conceptual design included the configuring of a nominal propulsion, propellant feed, pressurization, and structural

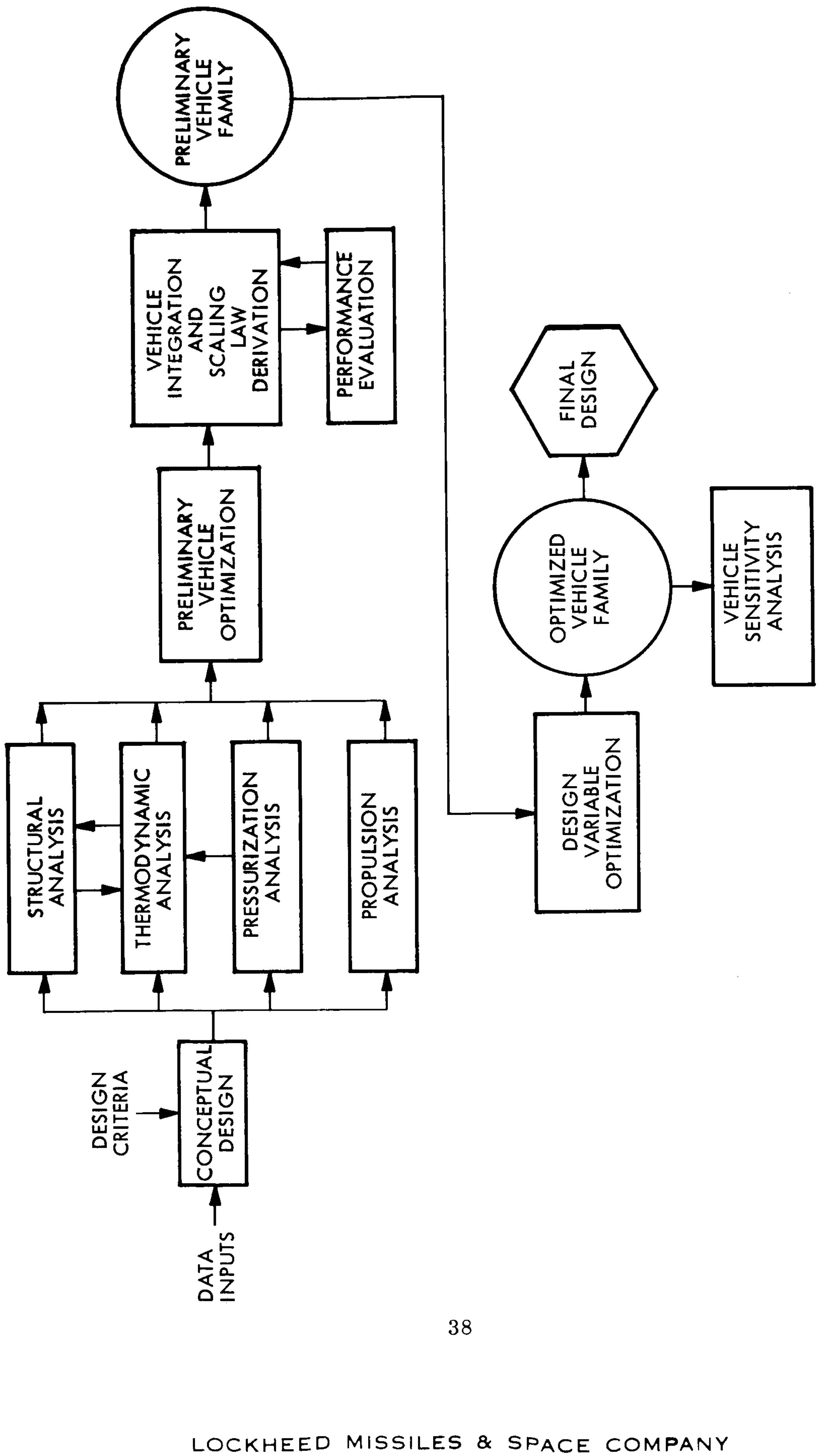


Fig. 7 Procedure for Stage Analysis – Task II

system for each propellant combination. The conceptual design provided the basis for the technical analysis.

2.1.2 Structural Analysis

The structural analysis consisted first of the development of parametric tank weights as a function of size, shape, and pressure. Aluminum tanks were used because of their wide applicability. Fiberglass tank supports with titanium end fittings were used to minimize structural heat leaks. The remaining structural elements, such as engine supports, internal structure, meteoroid shield, and all load-carrying members, were specifically designed for each application. The meteoroid shield consists of a dual-wall, foam-filled sandwich.

2.1.3 Thermodynamic Analysis

The thermodynamic analysis consisted of developing a thermal model of the vehicle, defining the external environment throughout the mission, selecting appropriate tank surface coatings, conducting a vehicle energy balance, and then optimizing the thermally sensitive parameters (coatings, insulation, tank pressure, tank dry weight, propellant boiloff, propellant initial temperature, etc.) to minimize total system mass. The thermal model consisted of a radiation resistance network, a conduction resistance network, and lumped capacitance nodes in order to accurately define the vehicle thermally. A heat-rate program was then used to assess the vehicle component surface temperature for various surface coatings as experienced during the various phases of the mission. The heat input into the tanks was then determined for each propellant for various insulation thicknesses using a thermal analyzer program. For vented systems, the combined effects of insulation weight and boiloff weight, with appropriate tradeoff factors, were used.

2.1.4 Pressurization Analysis

In the pressurization analysis, the appropriate pressurant gas, gas pressure, gas inlet temperature, and gas storage pressure and temperature were determined. The effect of the number of burns, volumetric requirements, and vent/nonvent decisions were assessed and fed back to the gas-state selection. A complete ullage and liquid-energy balance was performed, and the Epstein Correlation Program was used to determine the actual pressurant gas requirements.

2.1.5 Propulsion

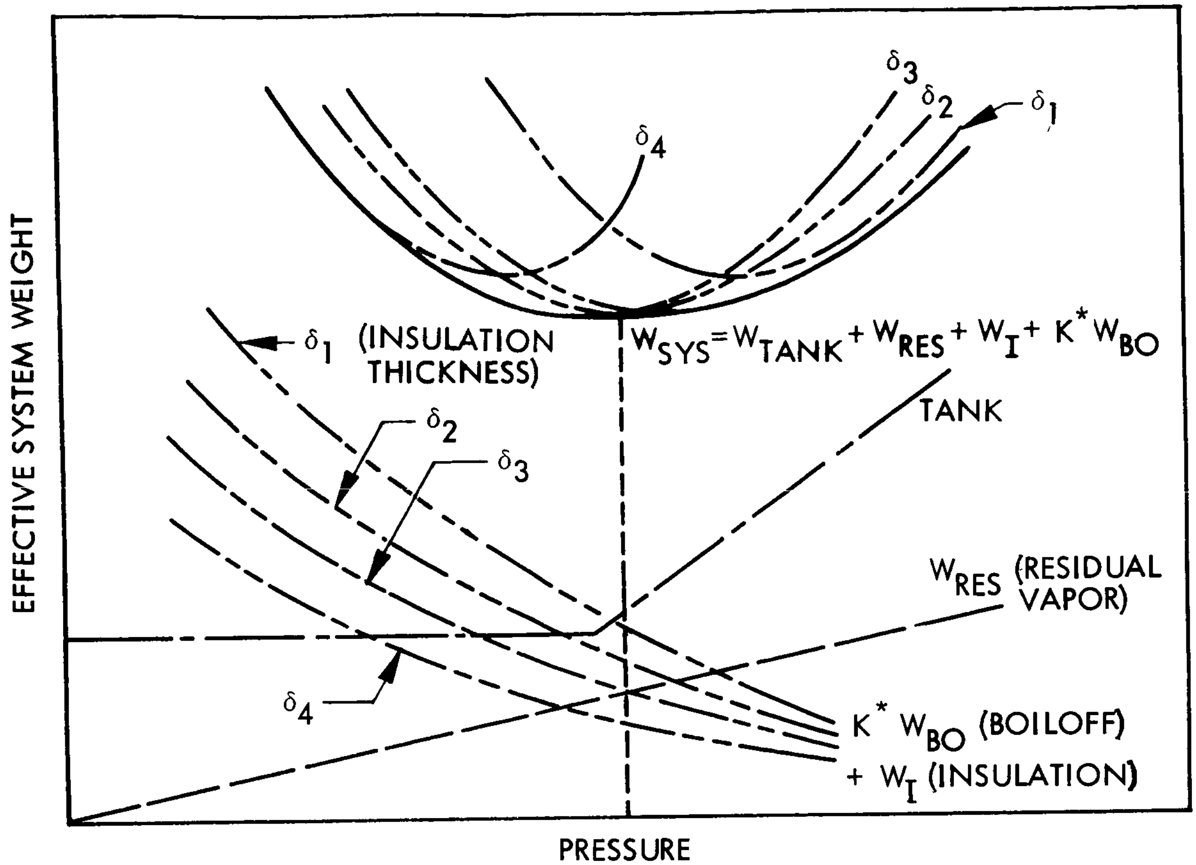
The propulsion analysis consisted of defining the propulsion requirements, sending this information to the supporting propulsion companies (Aerojet-General Corporation, Pratt & Whitney Aircraft, and Rocketdyne), and then assessing the performance, weight, and operational characteristics of each of the engine systems proposed by the propulsion companies. An engine system specification was then defined to meet the design criteria. An assessment of propulsion modes was then made for the mid-course trim and other corrective propulsion steps.

2.1.6 Preliminary Vehicle Optimization

The first step in the vehicle optimization consisted of a thermodynamic/structural optimization involving the propellant tank, insulation, residual vapor, pressurant, and boiloff. In this computation, the tank pressure for each propellant and the insulation thickness that yielded the lowest initial stage weight were defined. For tanks that required venting, the analysis was identical except for the insulation thickness. This step is shown in Fig. 8. All pressure-dependent systems were then completely defined, together with the insulation system.

2.1.7 Vehicle Integration

All of the pressure-dependent systems were combined with the pressure-independent systems in order to define a preliminary vehicle. The effect of changing the propellant loading was then determined by defining another point design. This yielded a propellant-sensitive scaling law that was used in the performance analysis.



*K = Factor to account for difference between launch mass and mass at start of burn

Fig. 8 Propellant System Optimization – Vented Tanks

2.1.8 Performance Evaluation

For these missions, the performance parameters were specified and the scaling law used to exactly determine the propellant loadings and inert weight fractions required to accomplish each mission.

2.1.9 Preliminary Vehicle Family

A preliminary vehicle family representing all propellant combinations was then synthesized. These vehicles were all defined with common groundrules and nominal engine systems so that a direct comparison could be made with all the peculiarities of each system normalized to an equal base.

2.1.10 Design Variable Optimization

For some cases, an assessment was made for each vehicle and each propellant combination to determine the effect of varying the chamber pressure, expansion ratio, and mixture ratio in order to further refine the performance optimization.

2.1.11 Optimized Vehicle Family

Again, a family of vehicles representing all the propellant combinations was synthesized. The difference now, however, was that rather than all of them based on common groundrules, each vehicle was developed to its utmost capability.

2.1.12 Vehicle Sensitivity Analysis

The optimized vehicle family was then perturbed by varying the following parameters:

- Mission Length
- α/ϵ Characteristics
- Meteoroid Flux
- Specific Impulse
- Propellant Initial Condition
- Insulation Conductivity
- Vehicle Orientation
- Venting Requirements
- Secondary Propulsion Steps

2.1.13 Final Design

Subsequent to all of these steps, a final design for each vehicle/propellant combination was defined with drawings, weight statement, operating characteristics, and descriptions.

Section 3
MARS ORBITER STAGE INVESTIGATION

3.1 MARS ORBITER INPUTS AND ASSUMPTIONS

The Voyager vehicle as defined by TRW was selected by the NASA committee to serve as the basis for a typical Mars Orbiter mission vehicle. This vehicle is shown in Fig. 9 and described in Appendix B. Design and analysis was performed only on the propulsion module of the vehicle in order to adapt it for the various propellant combinations. The selected mission consisted of the following:

- 1973 Mars Orbiter/Lander
- 205-day duration with 195-day interplanetary trip, and orbit trim after 10 days in orbit about Mars
- 6,950 ft/sec total velocity
- Parking orbit ascent mode to 100 nm with up to 90 min in earth orbit

To assess the vehicle performance and conduct the propulsion module analysis, a payload capsule of 5,000 lb and a bus of 3,143 lb were assumed. The following four propulsive steps were stipulated:

- First midcourse = 164 ft/sec at T = 3 days
- Second midcourse = 164 ft/sec at T = 165 days
- Orbit insertion = 6,294 ft/sec at T = 195 days
- Orbit trim = 328 ft/sec at T = 205 days

All propulsive steps were conducted with the bus as payload, and all but the last step were conducted with the capsule also as part of the payload. The nominal thrust of the primary engine was selected as 8,000 lb, and was used at this thrust rating only for the orbit insertion maneuver. For the other propulsive maneuvers, both a throttled main engine and a secondary engine were considered.

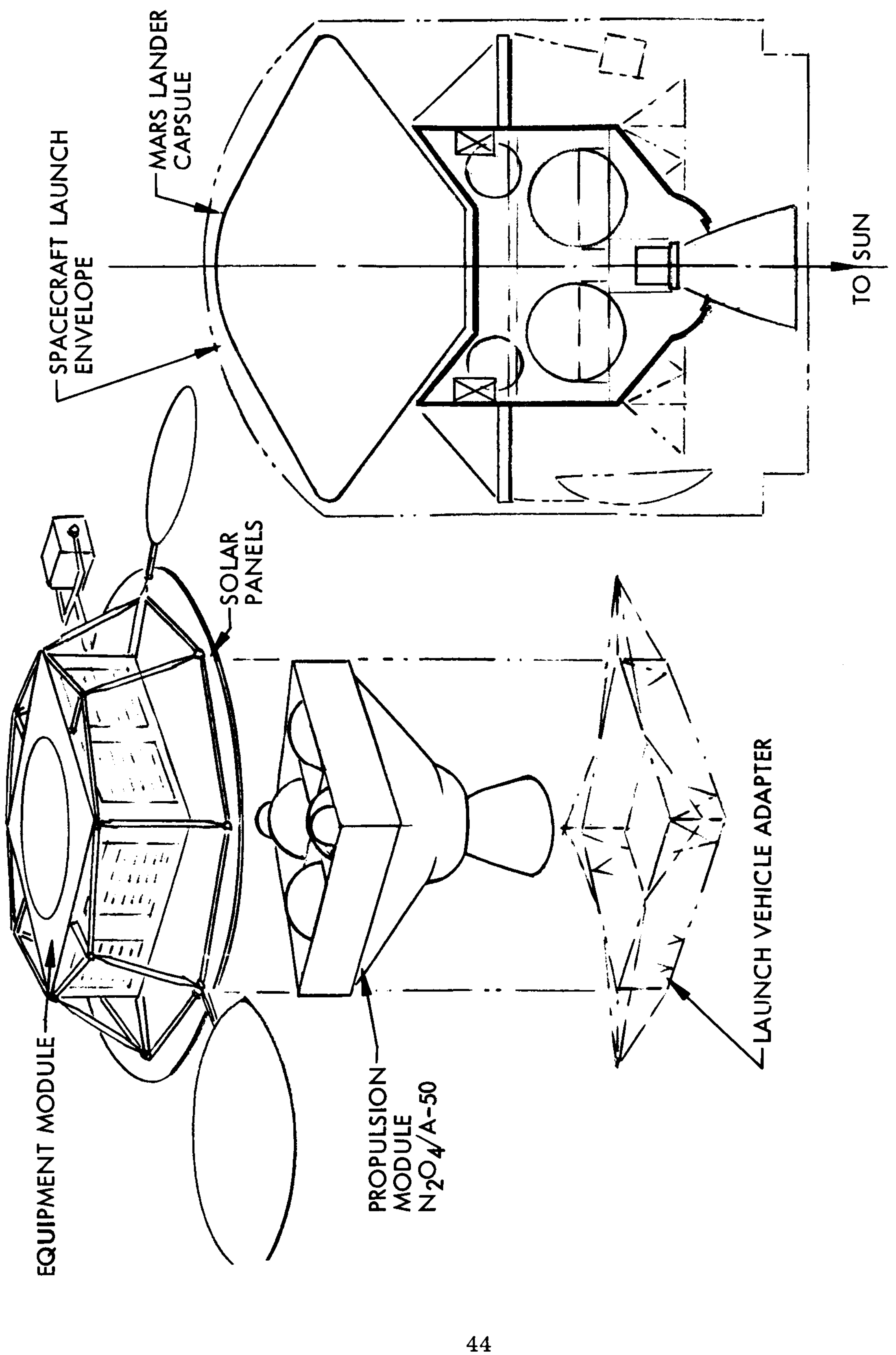


Fig. 9 Baseline Orbiter – TRW Voyager

The propellants used in this investigation included the following:

- F_2/H_2
- O_2/H_2 and O_2/H_2 subcooled
- $FLOX/CH_4$
- OF_2/B_2H_6
- F_2/NH_3
- ClF_5/MHF-5
- N_2O_4/A-50

3.2 MARS ORBITER DESIGN

Design concepts were developed for the earth storable, space storable, and cryogenic systems. Each class of propellants has specific thermal and configuration requirements necessitating alternate propulsion module designs. All concepts were developed according to a common set of criteria. The following criteria were used:

- Voyager baseline envelope and field joint, as defined in the TRW study
- Nominal ullage volume of 3 percent for earth storables, 7 percent for H_2, and 5 percent for all others
- Load factors included the following:
  - Launch: Axial at max load = 6 g; at rebound = -1.5 g
    Lateral = +1.5 g
  - Orbit insertion: Longitudinal = 0.75 g
  - Orbit trim: Longitudinal = 1.5 g (only if main engine is used at full thrust for orbit trim)
- Design factor of safety was 1.25 to ultimate stress at a zero margin of safety. Check for no yield at limit which equals 1.1 times maximum applied load.
- Meteoroids:
  - Flux model from Refs. 1 and 2 (revised)
  - Penetration model from Refs. 1 and 2 (revised)
  - Probability of no penetrations = 0.99
  - Propellant tanks are not to be used as part of the shield

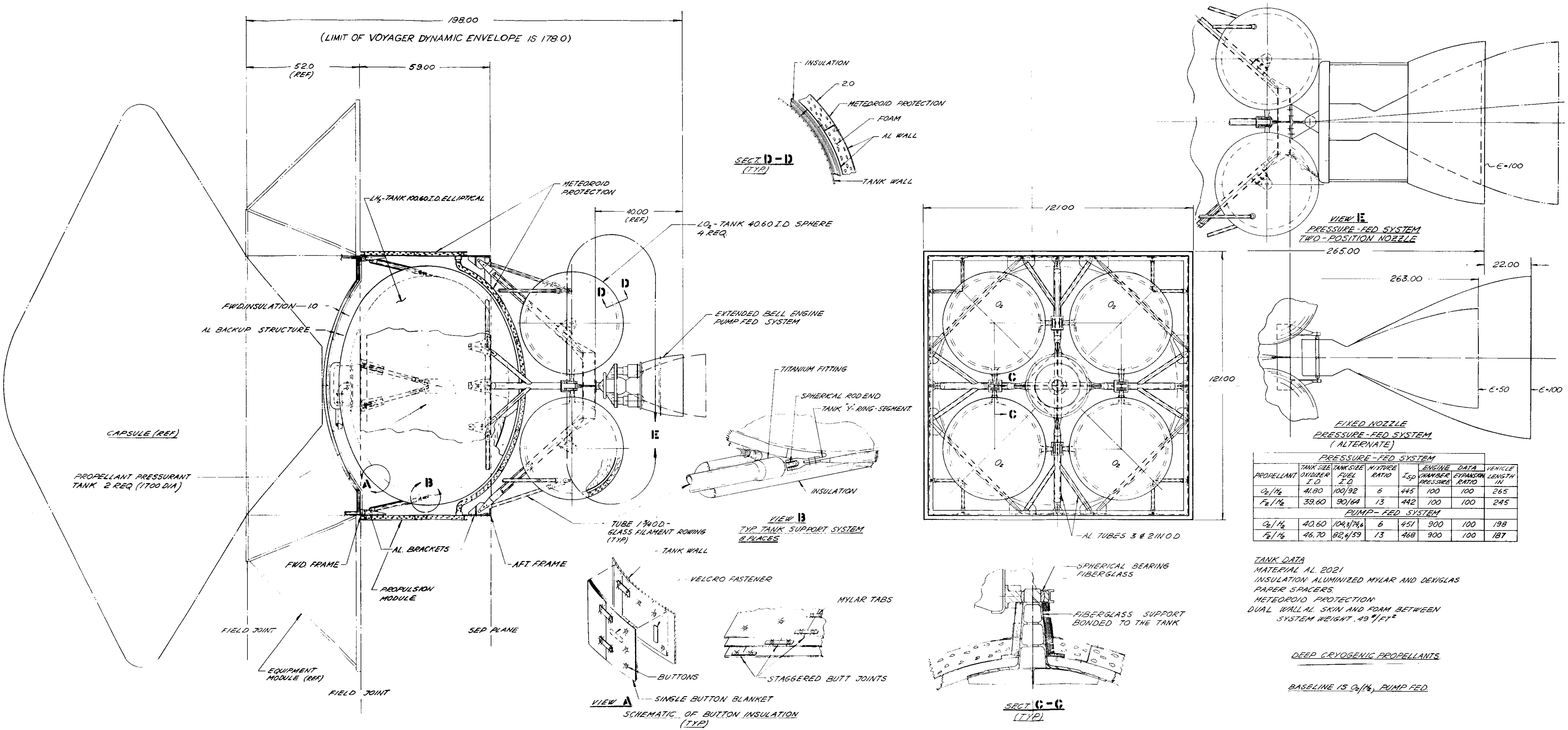
- Materials:
  - Tanks: use welded aluminum 2021 for all propellant tanks
  - Tank supports: low-heat-leak supports as appropriate, based on LMSC experience (probably fiberglass)
  - Insulation: double-aluminized Mylar with Dexiglas paper spacers; density = 4 lb/ft^3 and conductivity = 2.5×10^{-5} Btu/ft-hr-°R

3.2.1 Mars Orbiter Stage Concepts

The cryogenic concept used for the F_2/H_2 and O_2/H_2 propellant combination is shown in Fig. 10. The propulsion module has a square cross section 121 in. on a side. The length is propellant and engine-feed dependent. The propulsion module consists of two frames, forward and aft, and is connected by meteoroid panels, wrapped all around. These panels consist of a dual wall foam filled aluminum shield with 2-in. spacing. The inside wall serves as both structure and shield.

The engine is attached to a truss made of aluminum tubes and welded together. The engine loads are transferred into fittings on the aft frame of the propulsion module.

This concept has five tanks for the O_2/H_2 configuration, one elliptical H_2 tank, and four spherical O_2 tanks. For the F_2/H_2 configuration, there is one elliptical H_2 tank and two spherical F_2 tanks for the pump-fed configuration and four spherical F_2 tanks for the pressure-fed configuration. The elliptical tank is made of 2021-T6 aluminum alloy and has a $\sqrt{2}:1$ dome with the major axis varying with capacity requirements. The tank is externally insulated with multilayer insulation. The tank is supported by a four-point support system, as shown in Fig. 11. An aluminum alloy receptacle is welded into the tank to which the tank-supporting struts are attached. A spherical bearing is installed in the strut adjacent to the tank receptacle. This feature, together with a hinged joint that attaches the struts to the load-carrying forward frame, permits the tank to freely contract when filled without imposing bending stresses in the tank or support system. For accessibility, the tank has one 17.50-in. diameter manhole



PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO	VEHICLE LENGTH IN
PRESSURE-FED SYSTEM							
O_2/H_2	41.80	100/92	6	445	100	100	265
F_2/H_2	39.60	90/64	13	442	100	100	245
PUMP-FED SYSTEM							
O_2/H_2	40.60	104.3/74.6	6	451	900	100	198
F_2/H_2	46.70	82.6/59	13	468	900	100	187

TANK DATA
MATERIAL AL. 2021
INSULATION: ALUMINIZED MYLAR AND DEXIGLAS PAPER SPACERS.
METEOROID PROTECTION:
DUAL WALL AL. SKIN AND FOAM BETWEEN
SYSTEM WEIGHT .49#/FT²

DEEP CRYOGENIC PROPELLANTS

BASELINE IS O_2/H_2 PUMP FED

Fig. 10 Mars Orbiter Cryogenic Stage Details

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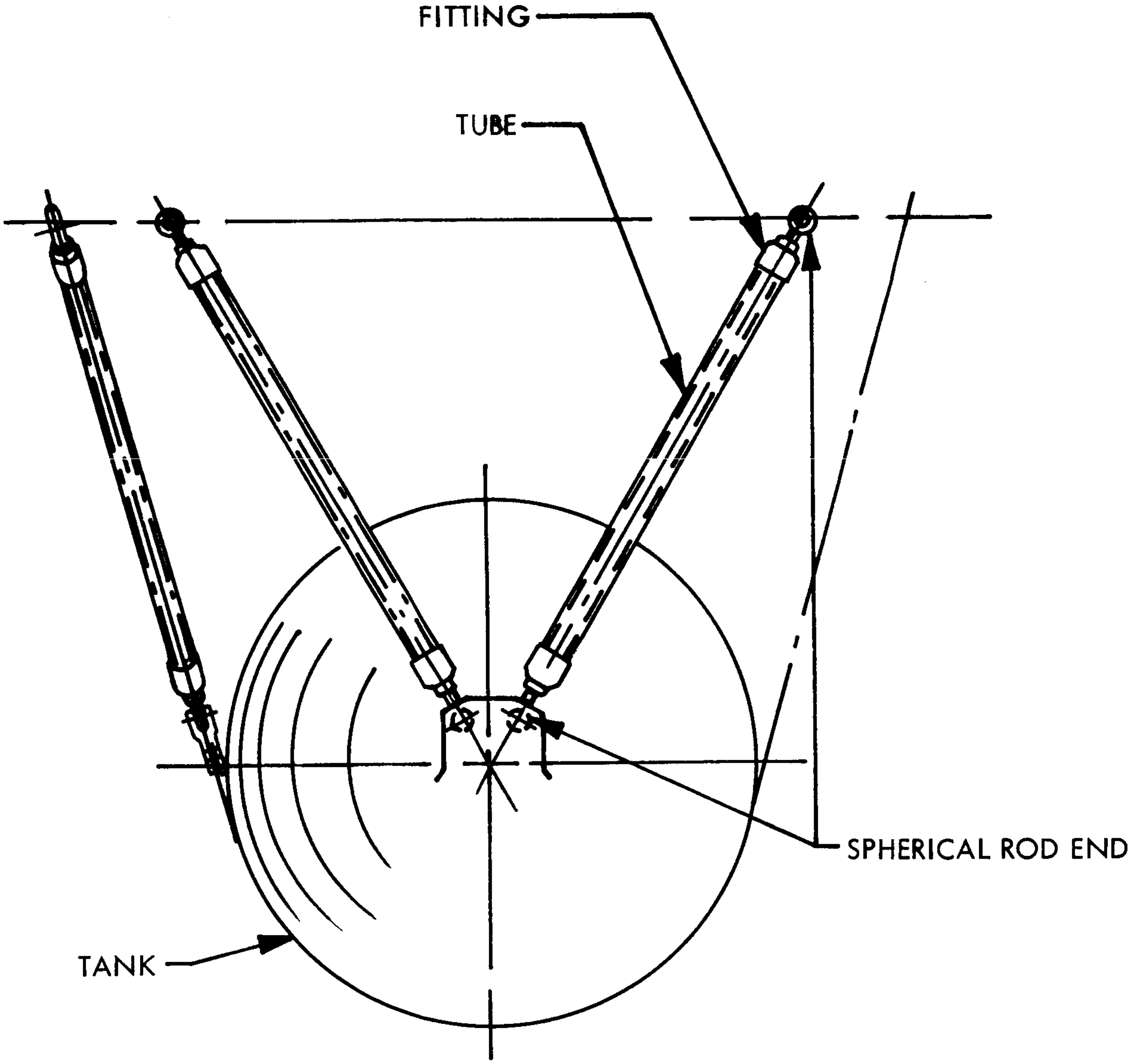


TYPICAL TANK SUPPORT SYSTEM

CONSISTS OF TUBE, TWO FITTINGS, AND TWO SPHERICAL ROD ENDS.
TUBE MATERIAL: GLASS FILAMENT WINDING
MAXIMUM DIAMETER: 1.75 OD
WALL THICKNESS: VARIABLE WITH TANK LOAD
FITTING MATERIAL: 6AL-4V TITANIUM
SPHERICAL ROD END: 6AL-4V TITANIUM

Fig. 11 Typical Tank Support System

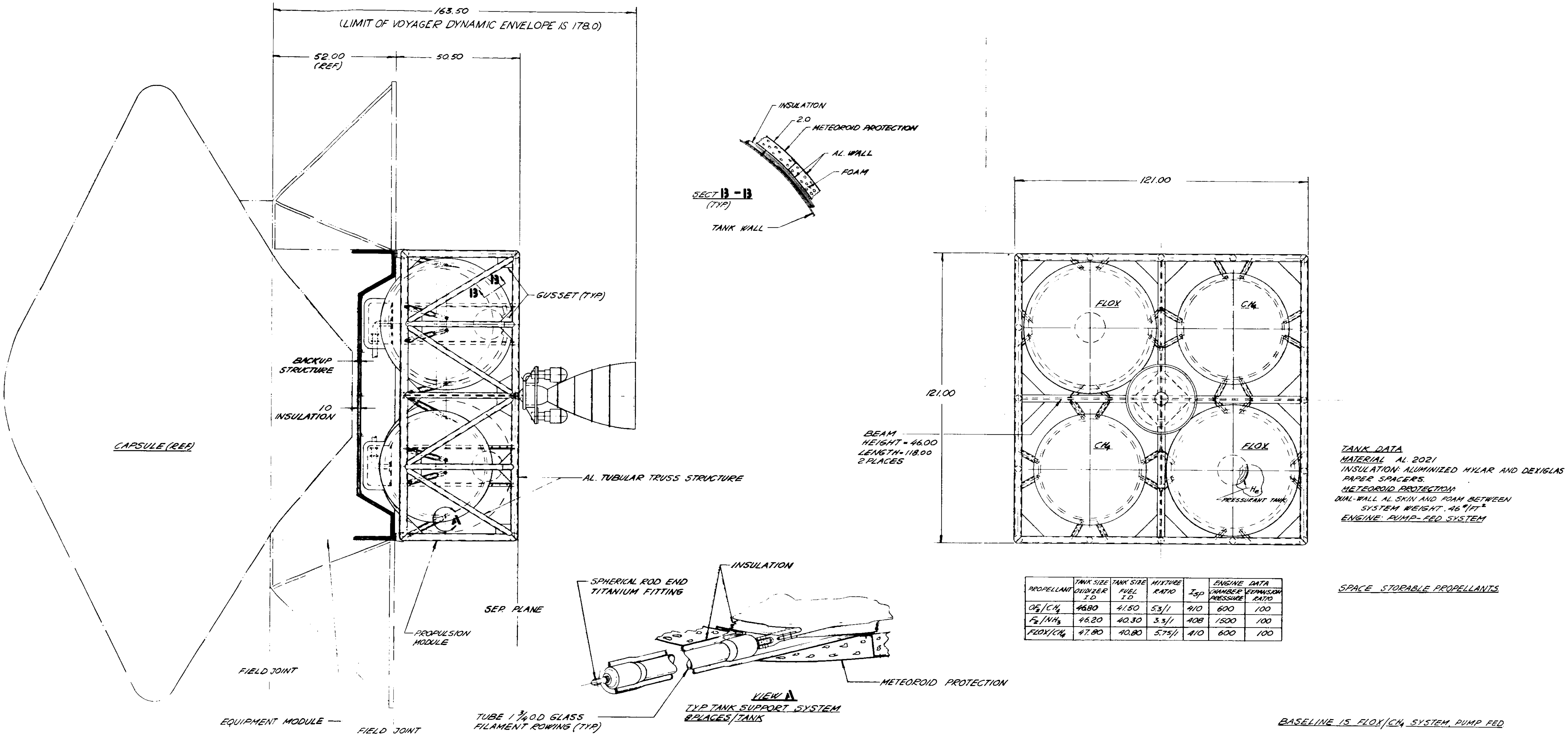
cover. The spherical tanks are also made of 2021-T6 aluminum alloy. For access, each tank has a 17.50-in.-diameter manhole cover. The spherical tanks are supported in a manner similar to the elliptical tank, with fiberglass struts and end fittings made of titanium. The tank diameter varies with capacity. Each tank is externally insulated with multilayered insulation, and outside of the insulation a meteoroid shield protects the tank. The shield consists of a foam-filled, dual wall with the walls spaced 2-in. apart. The meteoroid shields are made in two half spheres and are supported from the manhole and the aft boss. The aft dome of the elliptical tank is protected with a similar meteoroid system.

In the forward end between the propulsion and equipment module and just forward of the elliptical tank, a 1-in. superinsulation blanket is located for thermal isolation from the equipment bay and capsule.

The basic envelope constraint used by TRW in the Mars Voyager was exceeded for all cryogenic propellant combinations. The available length from the field joint was 178 in. All of the cryogenic systems exceeded that length, although two-position nozzles were used. This is especially true for the pressure-fed systems, as shown in Fig. 10.

The space-storable concept for FLOX/CH_4, OF_2/CH_4, and F_2/NH_3 pump-fed systems is shown in Fig. 12. The basic spacecraft is 163.5-in. long from the field joint between the capsule/equipment module and the aft end of the engine. The propulsion module uses an aluminum tubular truss structure, arranged as a square 121 in. on a side, with an attachment to two shear resistant beams arranged in a cruciform pattern that forms four identical compartments within the module.

Within each compartment a propellant tank is mounted using a four-point support system similar to the system shown in Fig. 11. Each propellant tank is spherical and is made of 2021-T6 aluminum alloy. For access, all tanks have one 17.50-in.-diameter manhole cover. Each tank is externally insulated with multilayer insulation,



PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO
OF_2/CH_4	46.80	41.50	5.3/1	410	600	100
F_2/NH_3	46.20	40.30	3.3/1	408	1500	100
$FLOX/CH_4$	47.80	40.80	5.75/1	410	600	100

Fig. 12 Mars Orbiter Space-Storable Pump-Fed Stage Details

and a meteoroid shield protects the tanks outside of the insulation. The shields are made in two half spheres, and are supported from the manhole and the aft boss.

The engine is mounted below the propellant tankage array and transmits the thrust load into the beams. The propulsion module matches the square form of the equipment module. An insulation blanket is located in the forward end between the propulsion module and equipment module and just forward of the spherical tanks. The space-storable concept for $FLOX/CH_4$, OF_2/CH_4, F_2/NH_3, and OF_2/B_2H_6 pressure-fed systems is shown in Fig. 13. The configuration is very similar to the pump-fed system except for the larger engine, which is equal to, or exceeds, the envelope limitations.

The earth-storable configuration for pump- and pressure-fed systems for N_2O_4/A-50 and ClF_5/MHF-5 propellants shown in Fig. 14 is basically the TRW configuration adapted to the study mission criteria. The basic concept of enclosing the volume around the tanks and integrating the structure, meteoroid shield, and insulation was carried over from the TRW design. Only the engine was left exposed. The module dimensions are given in Fig. 14.

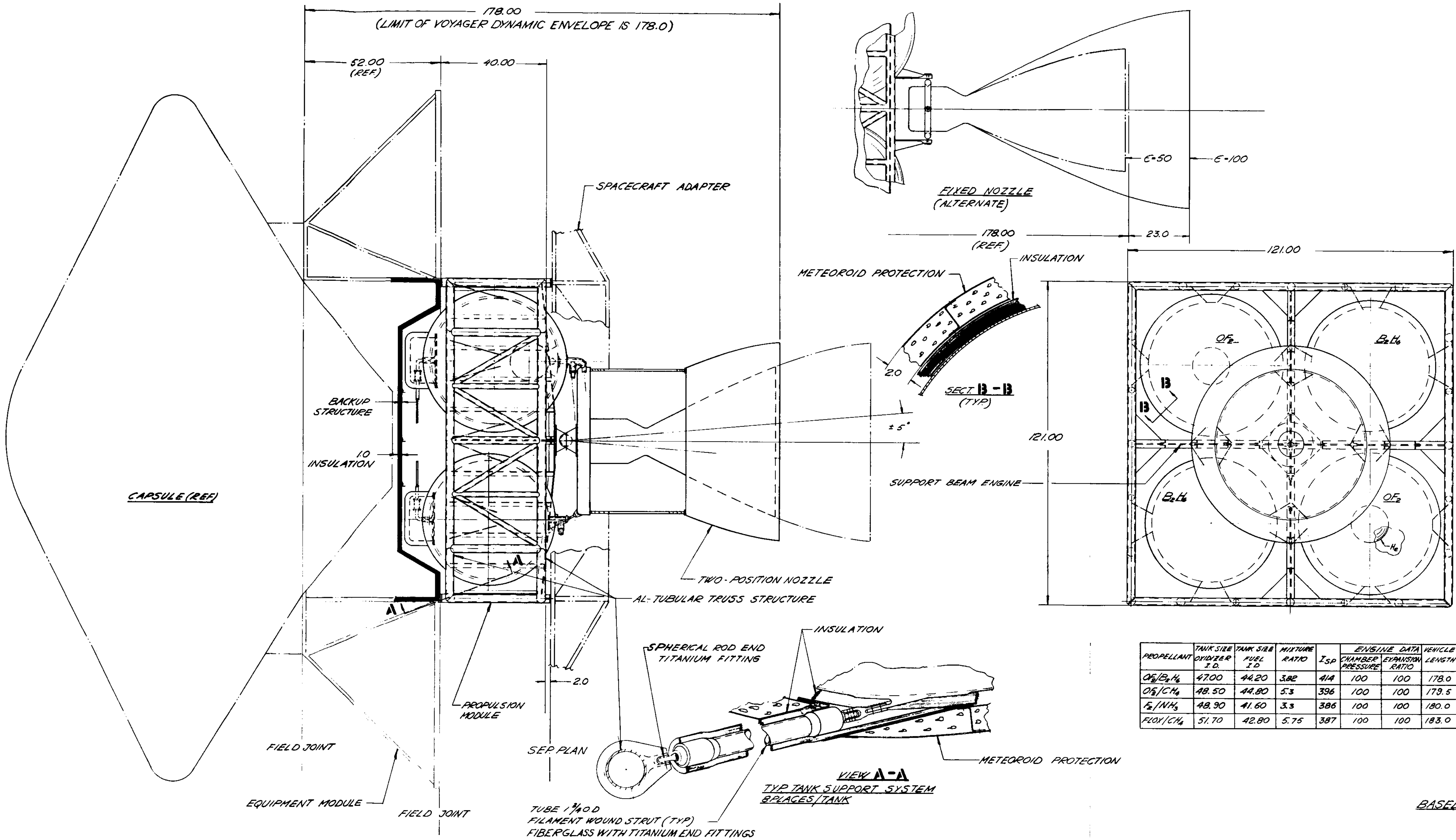
3.2.2 Mars Orbiter Stage Structural Analysis

Tanks. Parametric designs of propellant tanks for the Mars Orbiter were prepared for the following:

- Spherical tanks with diameters varying from 30 to 55 in.
- Ellipsoidal tanks ($\sqrt{2}$:1) with major axis diameters varying from 40 to 110 in.

All of these tanks were designed with the following requirements:

- Minimum skin thickness 0.040 in.
- 2021-T6 aluminum skin with +70°F allowables
- Tank pressures varied from 0 to 300 psi
- Manhole covers in all tanks, 17.50-in.-diameter
- Allowances made for local beef-up for support attachments and discontinuities



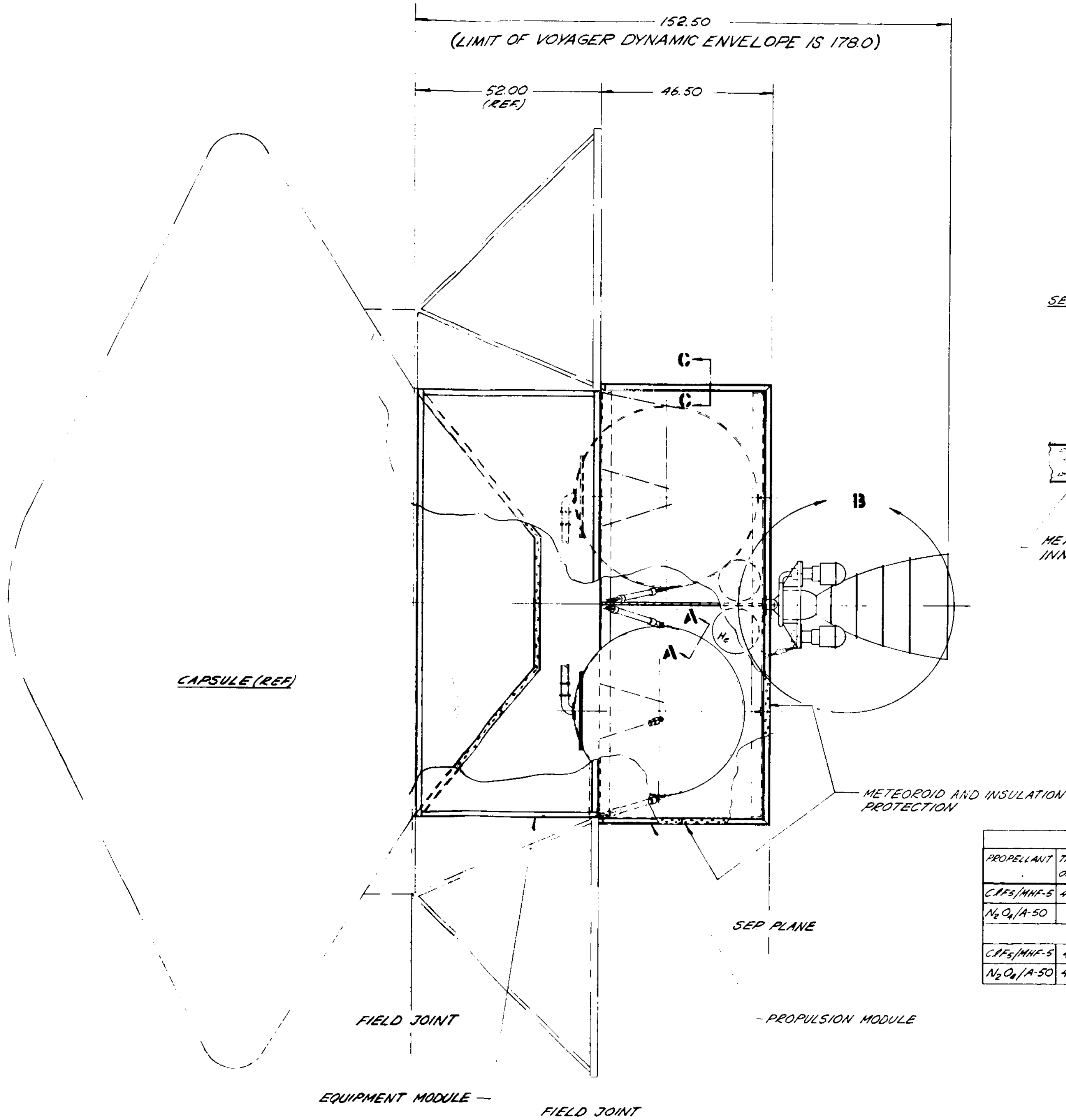
PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{SP}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO	VEHICLE LENGTH
OF_2/B_2H_6	47.00	44.20	3.82	414	100	100	178.0
OF_2/CH_4	48.50	44.80	5.3	396	100	100	179.5
F_2/NH_3	48.90	41.60	3.3	386	100	100	180.0
$FLOX/CH_4$	51.70	42.80	5.75	387	100	100	183.0

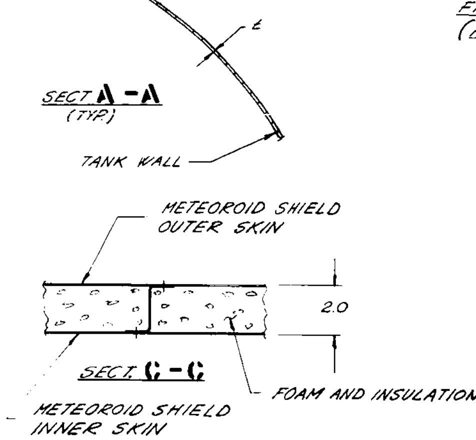
TANK DATA
MATERIAL AL 2021
INSULATION: ALUMINIZED MYLAR AND DEXIGLAS PAPER SPACERS.
METEOROID PROTECTION:
TWO AL SKIN AND FOAM BETWEEN
SYSTEM WEIGHT .48 #/FT²

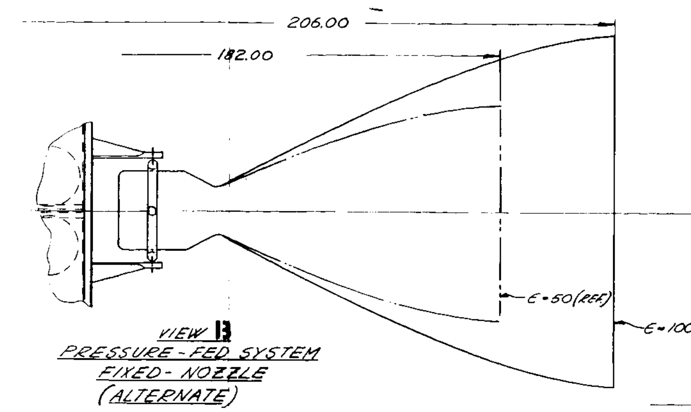
SPACE STORABLE PROPELLANTS

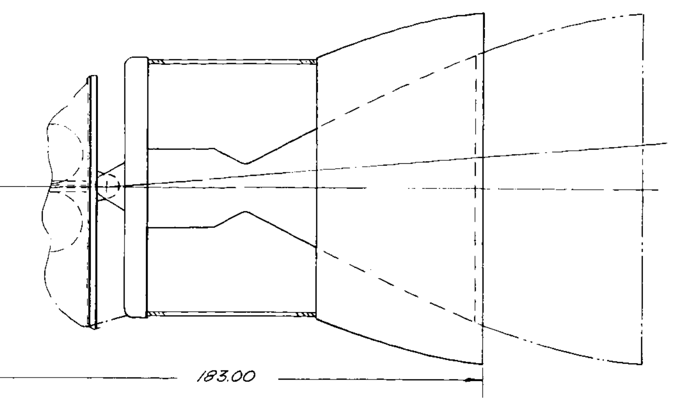
BASELINE IS OF_2/B_2H_6 SYSTEM, PRESSURE FED

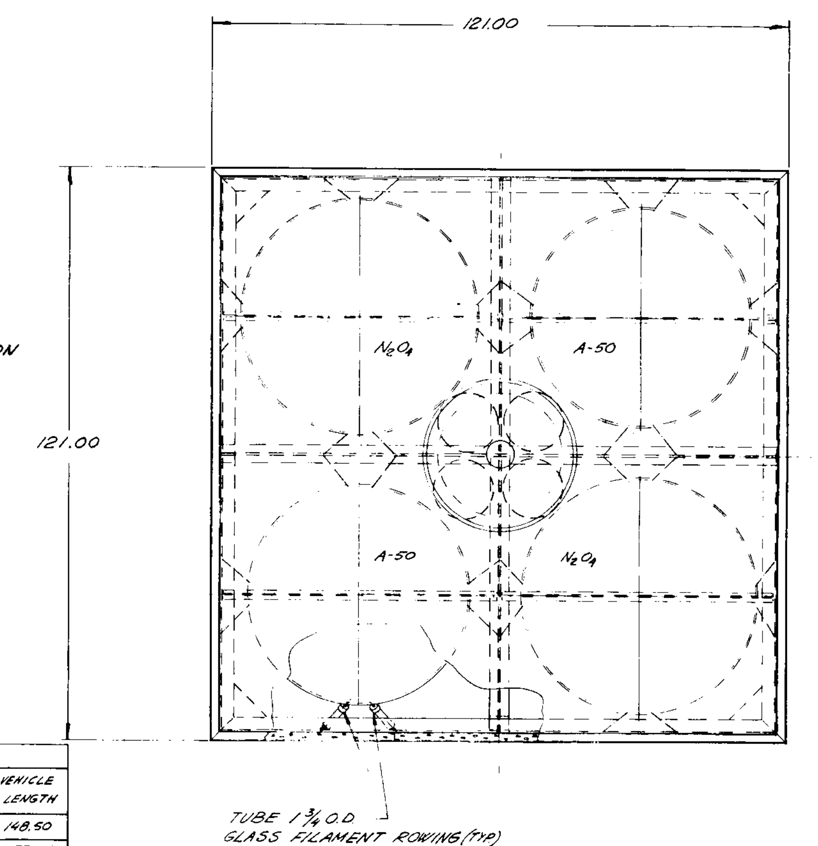
Fig. 13 Mars Orbiter Space-Storable Pressure-Fed Stage Details











TANK DATA:
MATERIAL AL 2021

METEOROID PROTECTION & INSULATION
DUAL WALL AL SKIN AND FOAM BETWEEN

PROPELLANT	TANK SIZE OXIDIZER	TANK SIZE FUEL	MIXTURE RATIO	I_{SP}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO	VEHICLE LENGTH
PUMP-FED SYSTEM							
ClF_5/MHF-5	44.40 I D	40.80 I D	2.40/1	342	750	100	148.50
N_2O_4/A-50	48.20	44.70	2.0	335	750	100	152.50
PRESSURE-FED SYSTEM							
ClF_5/MHF-5	45.50	41.80	2.4	330	100	100	179
N_2O_4/A-50	49.50	45.70	2.0	328	100	100	183

EARTH-STORABLE PROPELLANTS

BASELINE IS N_2O_4/A-50, PUMP FED

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Fig. 14 Mars Orbiter Earth-Storable Stage Details

Figures 15 and 16 show typical tank configurations for spherical and ellipsoidal shapes, respectively. Total tank weights as a function of tank pressure and for the range of tank sizes considered are shown in Figs. 17 and 18. The tank weights are independent of tank pressure up to the minimum gauge limitation, at which point they become very pressure sensitive.

Low-Heat-Leak Tank Support Design. An analysis of tank support systems for the Mars Orbiter was performed. The support systems consist of the structural support and associated insulation. The primary emphasis for the support struts is to achieve a minimum heat leak with maximum structural efficiency. It was determined that the most efficient system would be glass filament tubes as shown in Fig. 19. The struts consist of a single-filament-wound tube, which allows the use of higher design stresses. The outside diameter of the struts is 1.75 or 2.00 in. in all cases, with the wall thickness varied as the tank load varies for alternate propellants. With this design, the internal titanium end fittings are fixed in place on a salt mandrel. The tube is then continuously wound under tension around the mandrel and end fittings, making an integral structure. The external end fittings are then attached and the entire assembly is cured. After the cure, the salt mandrel is dissolved and washed out. The stress allowables for the filament wound tube are as follows:

- Tensile ultimate stress = 140,000 psi
- Safety factor = 1.5
- Modulus of elasticity = 5.4×10^6 psi

Meteoroid Shield. The basic meteoroid flux and penetration criteria are based on data specified by NASA and on Refs. 3 and 4. All designs were based on a probability of no penetration of 0.99 without using the tank surfaces as part of the shield. A dual wall, foam-filled, shield with aluminum face sheets spaced 2 in. apart was considered appropriate for this design. The shield yields an efficiency factor of 0.2. This can be defined as the total shield weight compared to a shield weight of a single sheet of aluminum providing the same protection.

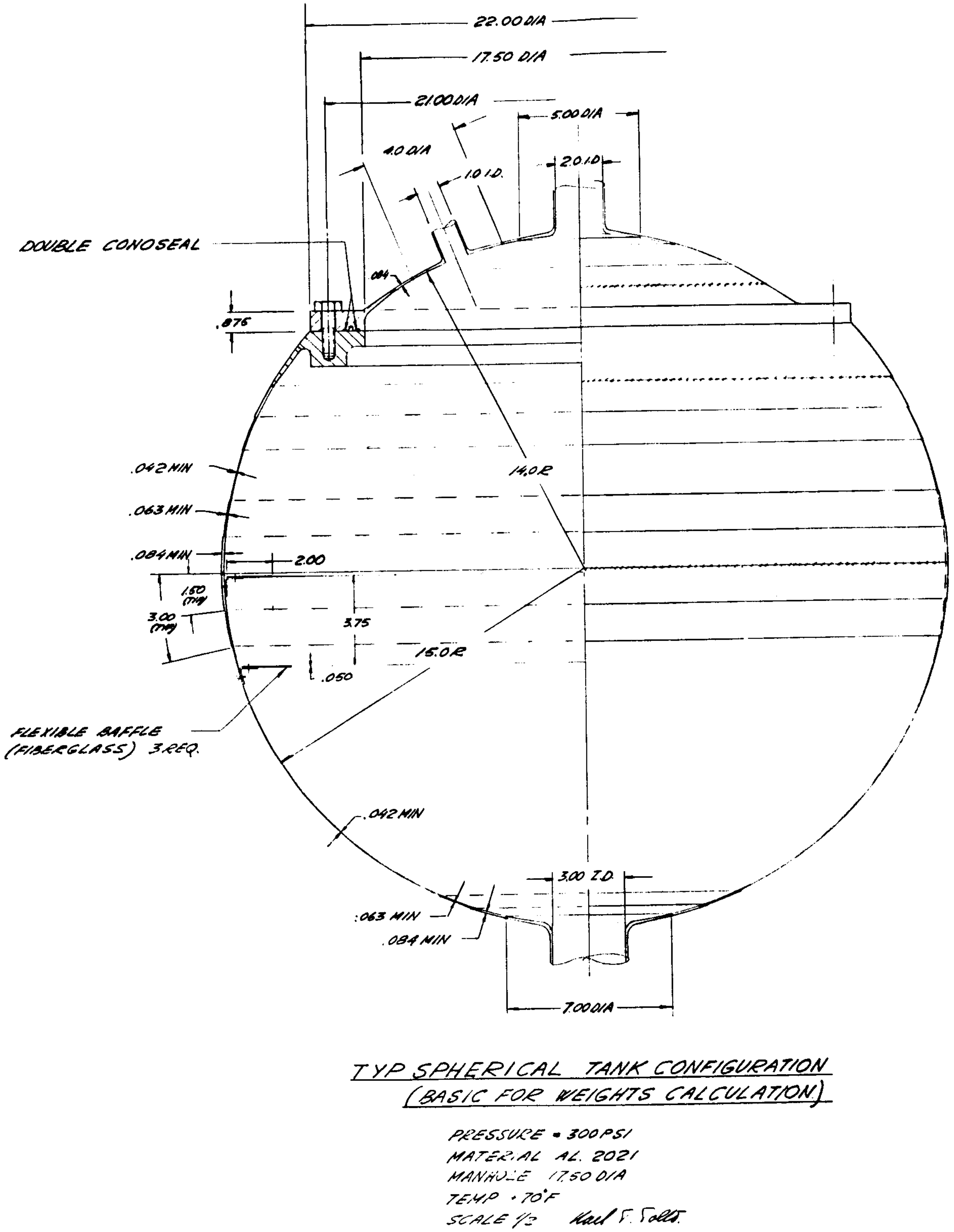

Fig. 15 Typical Spherical Tank Configuration

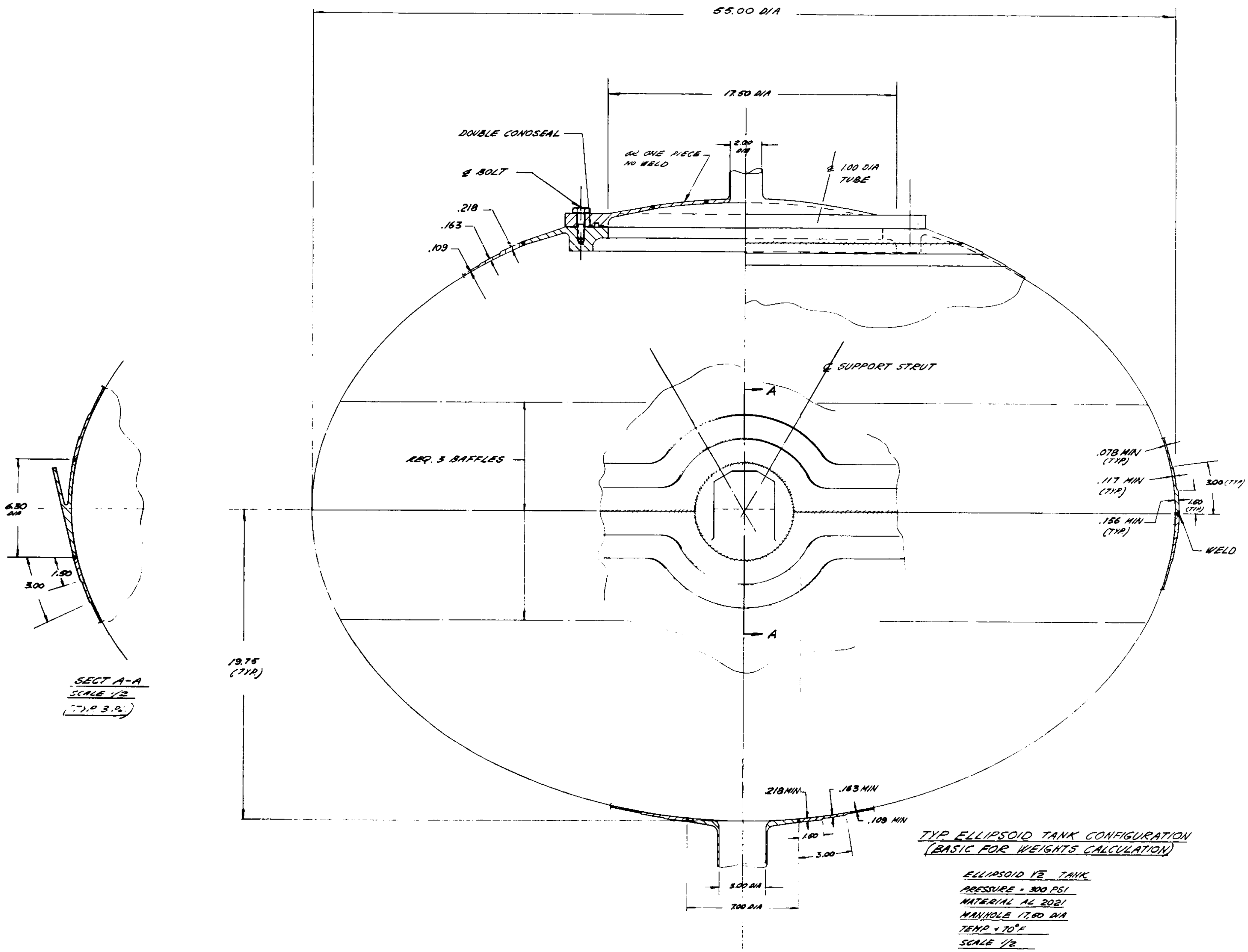

Fig. 16 Typical Ellipsoidal Tank Configuration

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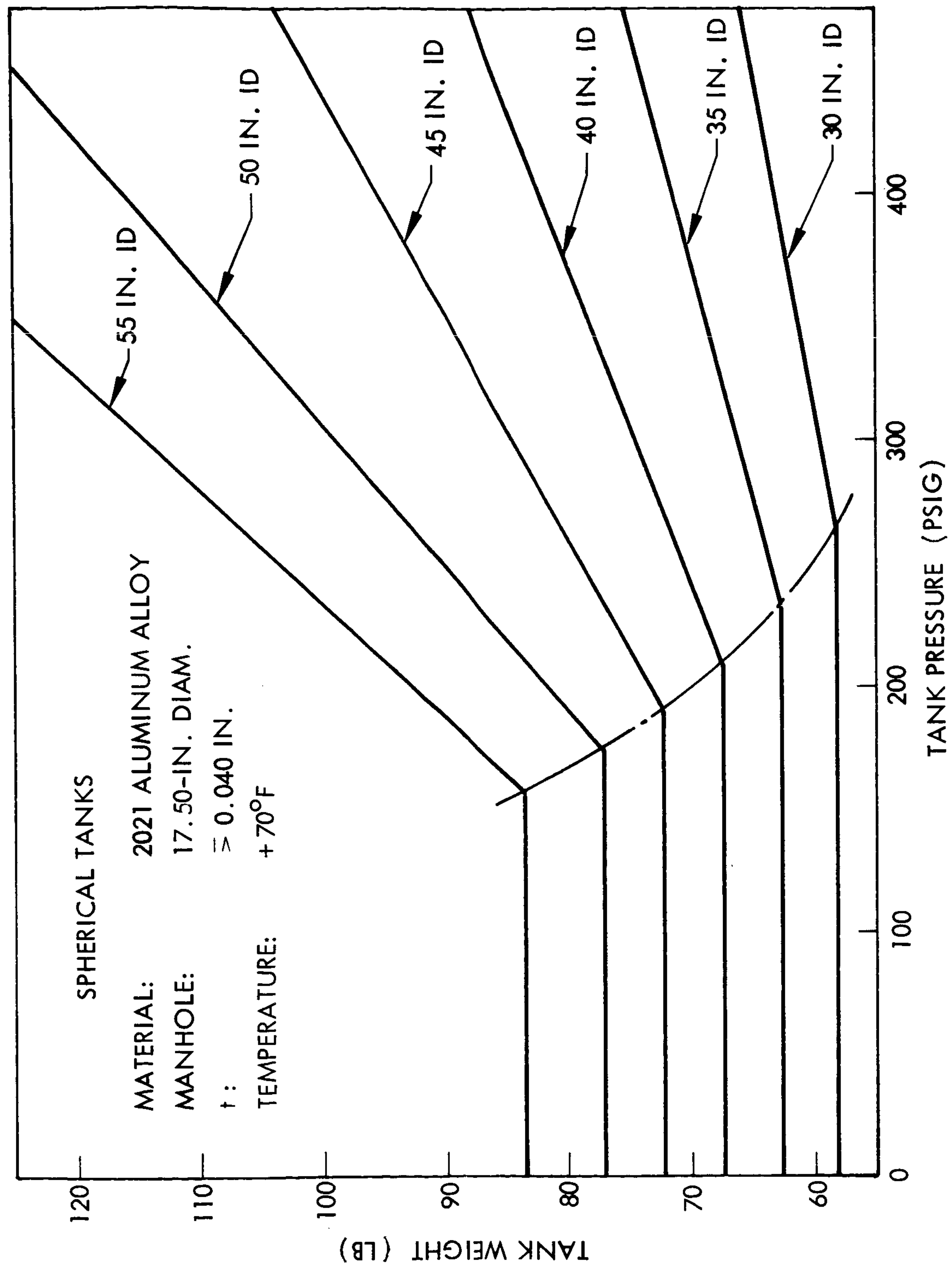


Fig. 17 Spherical Tank Weights

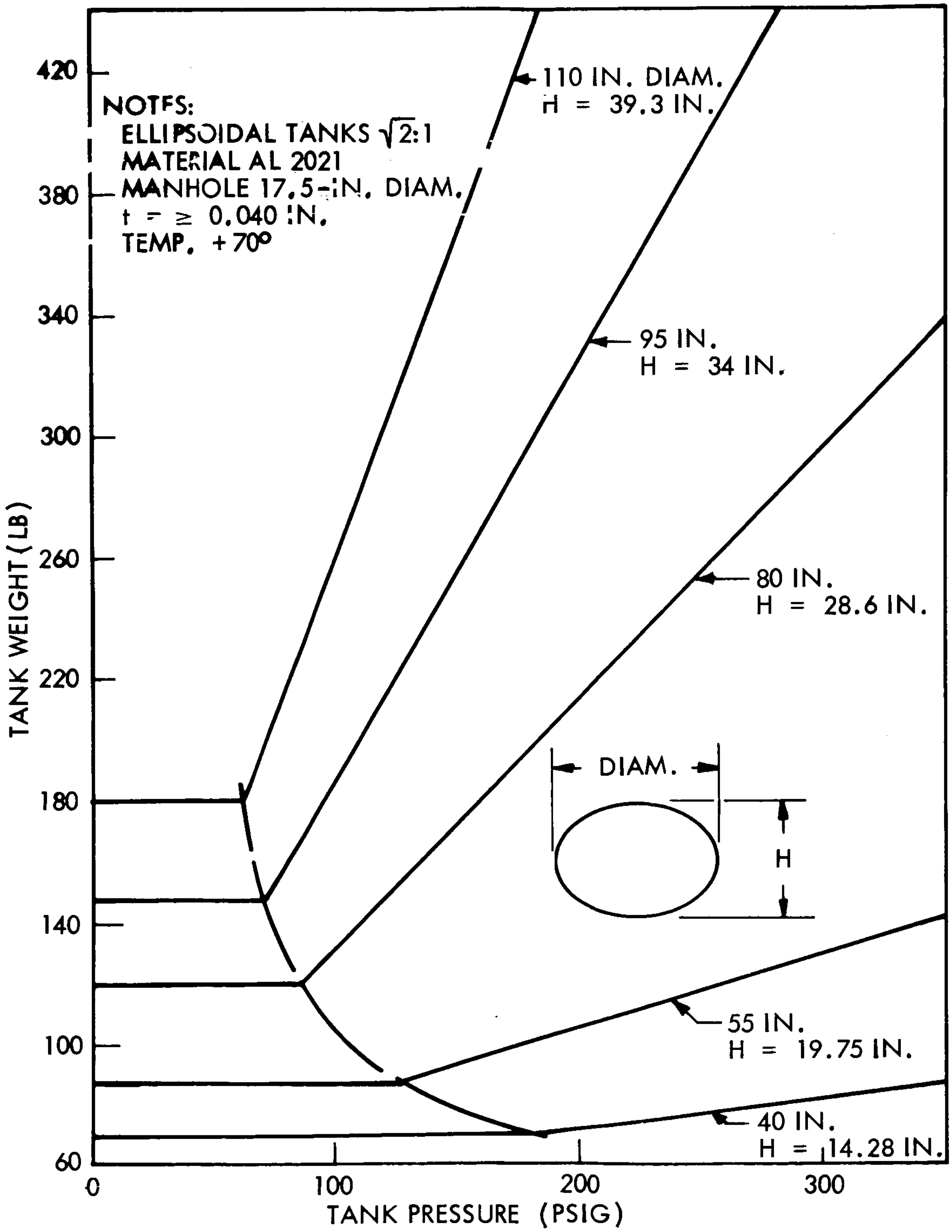


Fig. 18 Ellipsoidal Tank Weights

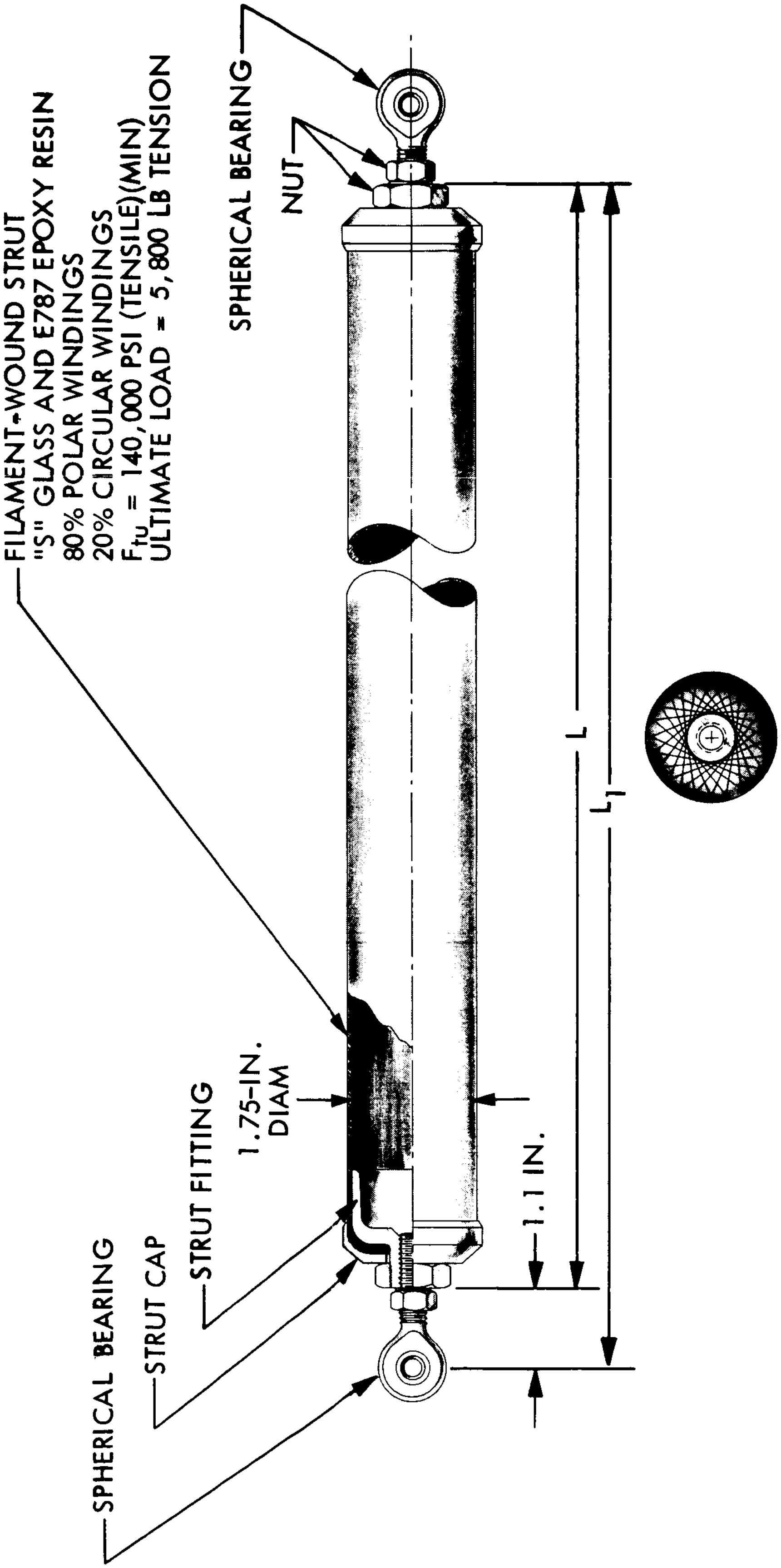


Fig. 19 Filament-Wound Strut – (Typical)

To apply these criteria to the vehicles, the realtionship of unit shield weight to the product of the exposed area and time is shown in Fig. 20. The total tank area is used as the effective area to be protected.

3.3 MARS ORBITER STAGE THERMODYNAMICS AND PRESSURIZATION ANALYSES

Thermodynamic analyses were conducted to determine the thermal control and pressurization system requirements for integration into the overall spacecraft design and computation of performance. The analyses were conducted in sufficient detail to provide valid performance comparisons among the candidate propellants. Heat transfer was computed to or from the propellant resulting from the external environment, other parts of the spacecraft, and from the pressurizing gas. Pressurization and thermal analyses could not be conducted independently because of the interaction between the pressurization system and thermal behavior of the propellant storage system. These integrated analyses established the optimum tank design pressure levels and insulation requirements that determine minimum weight of the system.

3.3.1 Thermodynamic Analyses Procedures

The thermodynamic analyses involved a number of discrete steps which resulted in the optimum values of thermal design parameters for each mission. The procedure is shown in Fig. 21, which traces the computation steps from input information to optimization output. Mission environment (i.e., trajectories, orbital altitudes, velocity, and orientation) definition allowed computation of heating rates. Thermal mathematical models were developed based on definition of the spacecraft configuration, equipment, and structural detail. A thermal analyzer computer program was then used to compute parametric temperature distributions and propellant heating data. With definition of tank configurations, duty cycles, and engine requirements, the modified Epstein pressurization correlation was used to compute collapse factors that influence pressurant gas requirements. These parametric data, together with the propellant heating data obtained with the thermal analyzer program, were used

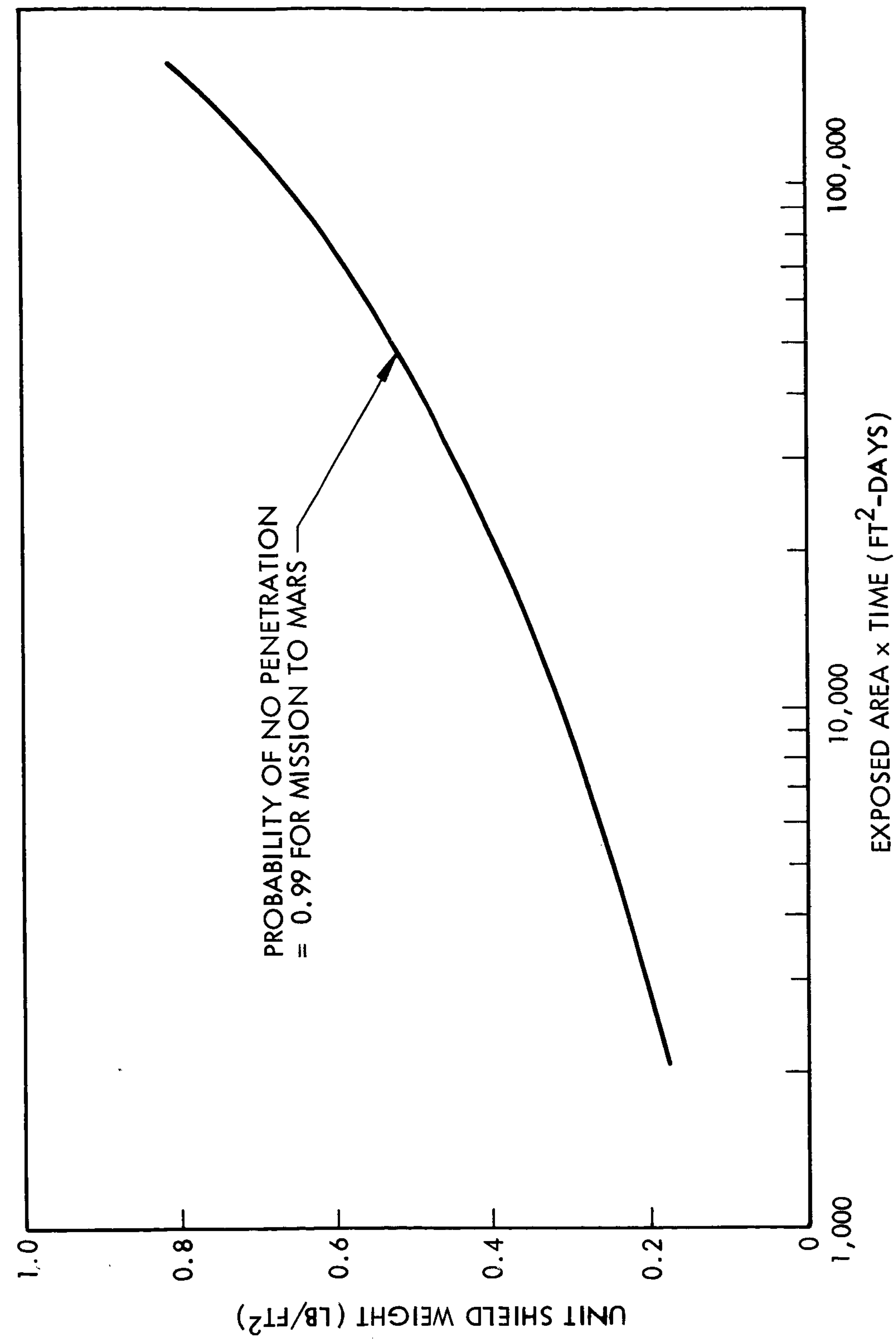


Fig. 20 Meteoroid Shield Unit Weight

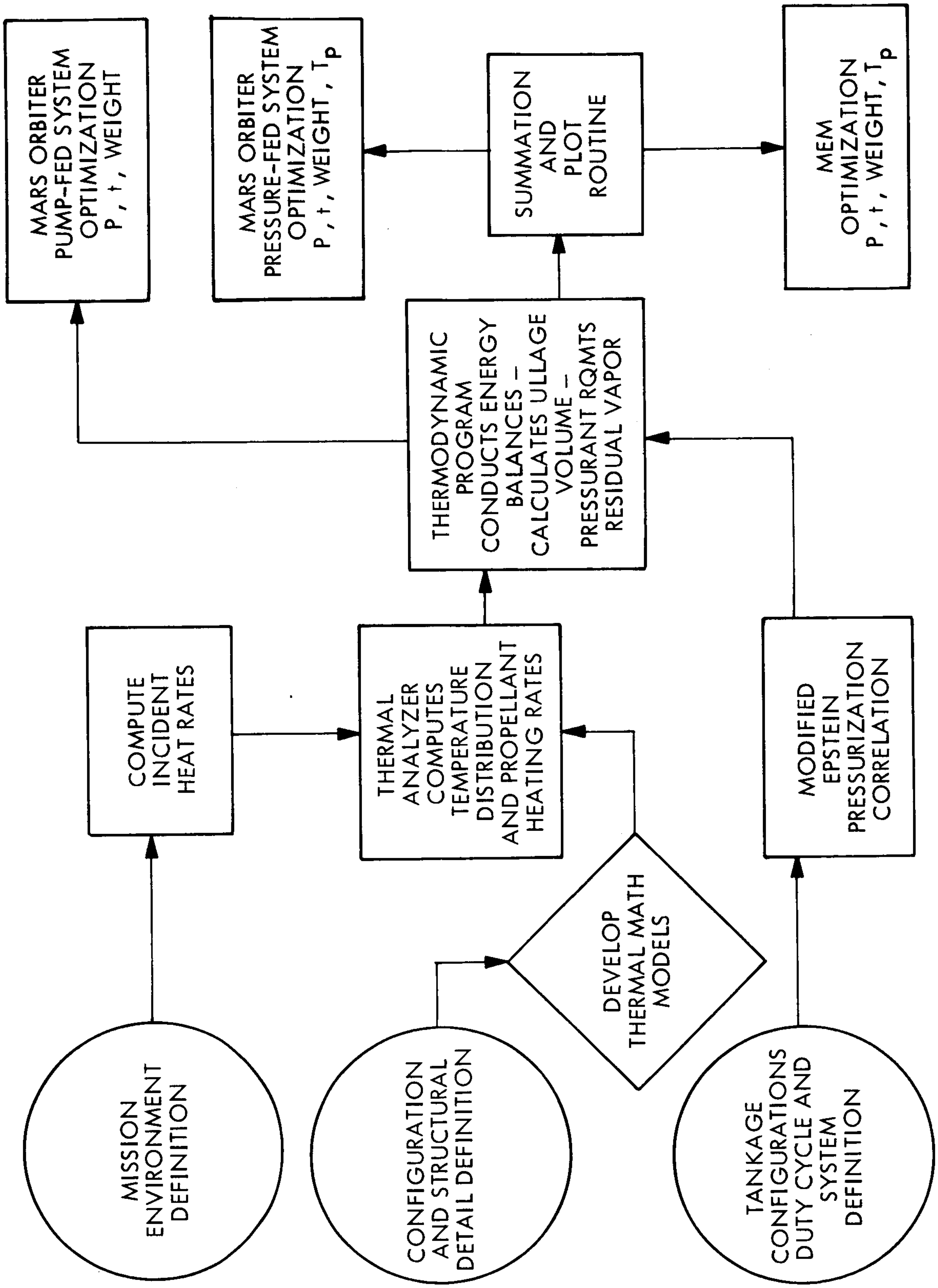


Fig. 21 Thermodynamic/Pressurization Optimization Procedure

in the Thermal-Pressurization program to compute insulation, pressurant system, tank, and residual vapor weights as a function of tank pressure, insulation thickness, and pressurant inlet temperature. A summation and plot routine was then used to plot total system weight as a function of an independent variable, usually tank pressure and/or insulation thickness.

Mars Orbiter Thermal Models. Models of two basic configurations were developed for the Mars Orbiter. One represented the design having a large ellipsoidal tank for hydrogen and spherical tanks for the oxidizer and the other represented the four-sphere configuration used for all other propellants.

The thermal models represent the physical elements of the spacecraft by an analogous electrical network consisting of conduction and radiation resistors and lumped node thermal capacitances. The thermal network can be used for analysis with external and/or internal heating rates. The total number of nodes and nodal distribution were selected to provide heat rates of sufficient accuracy to adequately evaluate the propellant response. The radiation portion of the network was developed by computing detailed geometrical view factors and specifying surface properties. The conduction portion of the network accounts for conduction through and around the vehicle structure, tank walls, insulation, and along feed, fill, vent, and pressurization lines. The heat inputs to a typical propellant system are shown in Fig. 22.

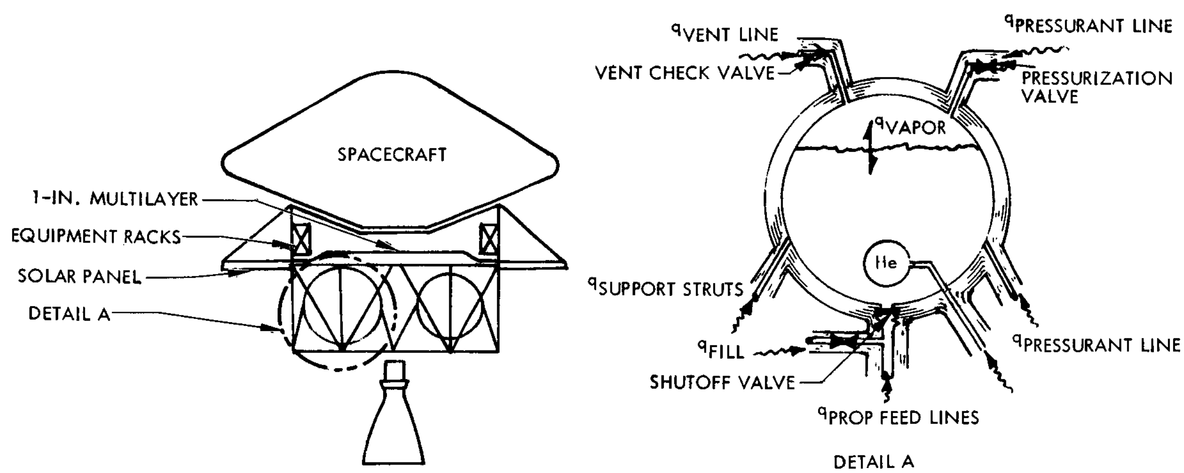


Fig. 22 Typical Propellant Tank Heat Inputs

Mars Orbiter Environment. The environment for the spacecraft was determined as a function of the mission sequence and trajectory. The primary energy source during the Mars transit is the sun. Solar flux density was computed as a function of time for specific trajectories, assuming the flux varied inversely with the square of the distance from the sun (Fig. 23a). Heating of the Mars Orbiter due to planet emission and abledo while in both earth and Mars orbits was found to be negligible.

Mars Orbiter Propellant Heating. The thermal analyzer program was used to compute temperature for all nodes of the thermal models and to compute heat flow into the propellant. The analysis was parametric, in that propellant heating rates were computed as a function of insulation thickness for boundary conditions imposed by surface finish properties and vehicle orientation. Each propellant was assumed to be at its normal boiling point at liftoff. The thermal capacitance of the propellant was computed based on the best specific heat and enthalpy data available for both liquid and vapor, and is discussed Volume III. All studies conducted assumed nonvented tanks except for hydrogen, where both vented and nonvented tanks were analyzed.

The external environmental heating rates (solar flux) decrease with time in the Mars transfer phase; therefore, temperatures of the spacecraft and the propellant heating rates decrease with time. Propellant heating rates also decrease because of a decreasing temperature drop across the insulation caused by the increasing propellant temperature. To handle this transient condition, this mission phase was divided into five equal time segments and each was analyzed as a steady-state condition. This approach gives a conservatively high total heat input. The propellant temperature response resulting from environmental heating is shown for F_2 tanks in Fig. 23b.

Mars Orbiter Pressurization. Helium pressurization systems were selected for all propellant tanks except for the pump-fed Mars Orbiter with hydrogen, which used a gaseous hydrogen bleed system. Helium was selected as the most applicable system for all propellants, and although it was not always the least-weight system, it provided a good comparison among propellants. The helium was assumed to be stored at the

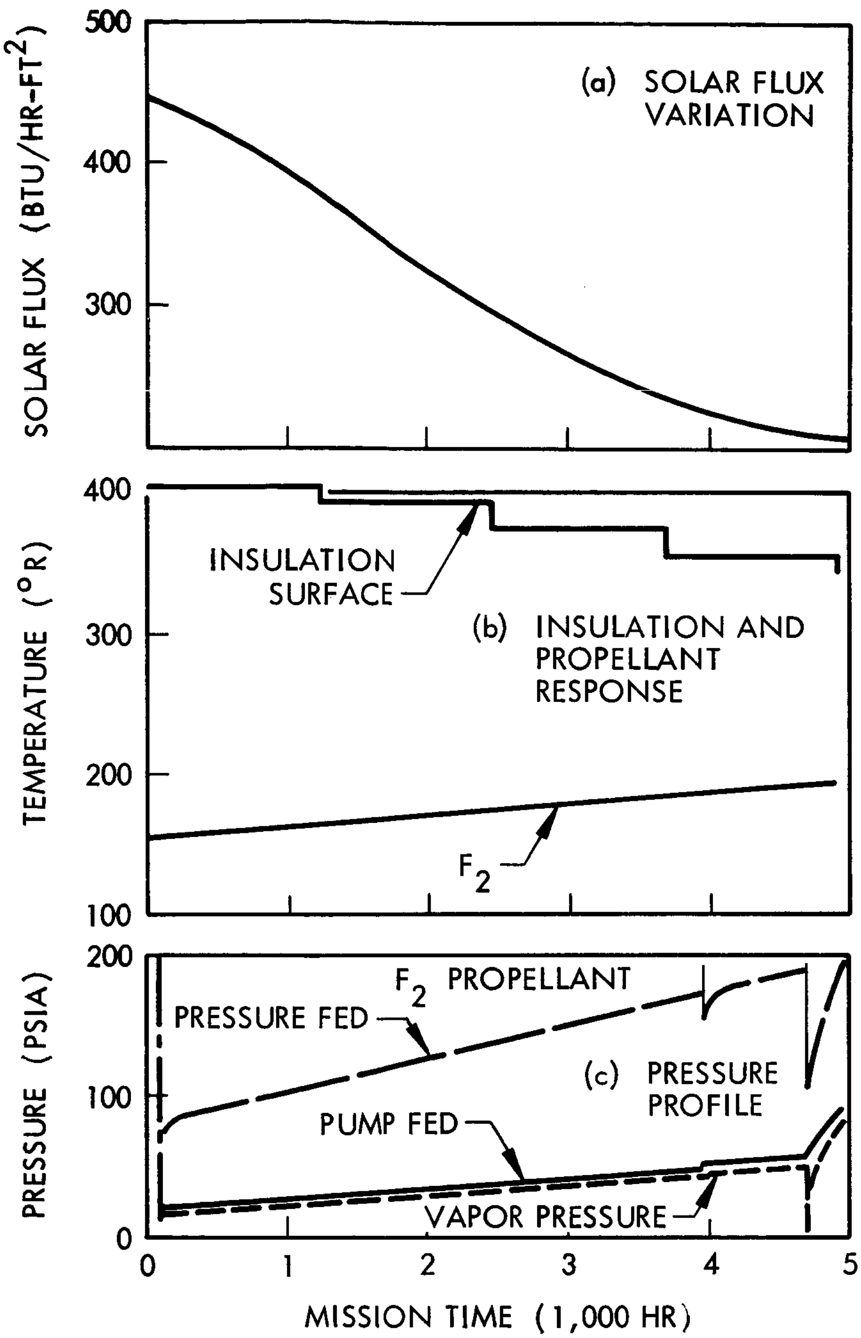


Fig. 23 Typical Mars Orbiter Environment Effects

highest saturation temperature of the colder of the oxidizer or fuel for any system. The appropriate propellant tank volume was increased to include the volume of the pressurant storage sphere.

Significant differences exist in the pressurization systems for the pump- and pressure-fed Mars Orbiter systems, as shown by the typical F_2 pressure profile of Fig. 23c. Therefore, details of the pressurization system and analyses were included under the individual system studies.

3.3.2 Mars Orbiter System Analysis – Thermodynamics and Pressurization

The Mars Orbiter mission consisted of a short earth-orbit phase (less than 90 min), a 195-day Mars transit, and 10 days in Mars orbit. The baseline mission duty cycle includes midcourse corrections at 3 days and 165 days from earth, an orbit inject burn at 195 days, and the orbit trim burn at 205 days. Studies were also conducted assuming that the orbit trim burn was accomplished with a secondary propulsion system.

Orientation of the spacecraft during the transit phase was assumed fixed relative to the sun with either the propulsion system tanks exposed or with the capsule exposed. The baseline orientation is sun on the tanks.

External thermal control surface properties were selected to give maximum performance. The lowest ratio of α/ϵ attainable (Optical Solar Reflector, OSR) was used for the cryogenics and space storables. Values that give tank surface temperatures within the liquid propellant temperature range were selected for the earth storables. Sensitivity analyses have been conducted to determine the effect of using thermal control surface properties other than the baseline values.

Mars Orbiter Pump-Fed Systems. Assumptions applicable to the pump-fed systems are as follows:

- Net Positive Suction Pressure (NPSP) = 4 psia
- Fixed ullage volume
- Helium pressurization system — gas injected at the propellant saturation temperature for precharge and expulsion
- No engine heat soak-back
- Idle mode start
- Thermal equilibrium between liquid and ullage was assumed between each burn

Insulation, pressurant, residual vapor, and tank weights were computed as a function of operating pressure. A minimum propellant tank gage of 0.040-in. was assumed This gives constant values of tank weight up to relatively high pressure levels.

Optimum values of tank pressure and insulation thickness were determined by plotting system weight as a function of pressure (Fig. 24). The optimum pressure and insulation thickness and the corresponding tank, vapor, and pressurization system weights were obtained for the point at which the system weight is a minimum.

Results of the thermal optimization study for the pump-fed system are shown in Table 11. The optimum operating pressure and insulation thickness and the corresponding total weights for a single propellant tank system of tank, vapor, insulation, and pressurization system for the baseline pump-fed system are shown on the left hand side of the table.

Mars Orbiter Pressure-Fed Systems. Analysis of the pressure-fed systems was conducted using the complete computerized methods previously described. Because relatively high tank pressures lead to substantial system weights, a detailed and accurate analysis was necessary to make an accurate comparison between propellants.

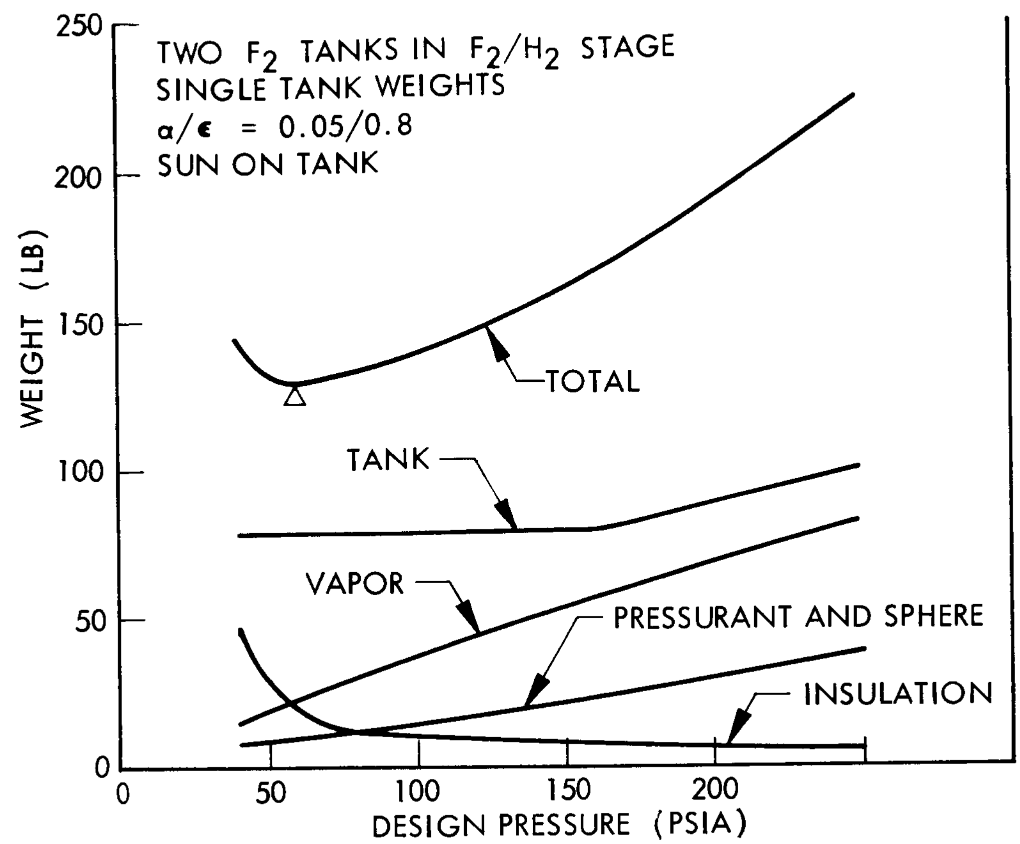


Fig. 24 Typical Pump-Fed Mars Orbiter Optimization

Assumptions applicable to the pressure-fed systems are as follows:

- Chamber pressure, P_c = 100 psia
- NPSP = P_c + feed system pressure drop; or P_c + saturation pressure, whichever is greater
- Heated helium pressurization system
- Ullage volume was adjusted to accommodate liquid expansion and helium compression
- No engine heat soak-back
- Idle mode start
- Thermal equilibrium between liquid and ullage was assumed prior to each burn

Table 11

MARS ORBITER THERMODYNAMIC OPTIMIZATION

Propellant	Pump-Fed System			Wt of Single Tank (lb)	Pressure-Fed System				Wt of Single Tank (lb)
	P(a) (psia)	T(b) (in.)	α/ϵ Ratio		P (psia)	T (in.)	Pressure Drop (psi)	α/ϵ Ratio	
F_2	40	1-1/8	0.05/0.08	116	190	1-1/4	90	0.05/0.80	112.7
H_2	130	4-5/8	0.05/0.80	495	198	4	90	0.05/0.80	805.3
O_2	58	3/4	0.05/0.80	107	195	1-1/4	70	0.05/0.80	123.6
H_2	96	4-5/8	0.05/0.80	725	175	3	70	0.05/0.80	1303.8
FLOX	57	1-1/2	0.05/0.80	137	195	1-1/2	95	0.05/0.80	204.8
CH_4	107	3/4	0.05/0.80	87	224	1-1/8	95	0.05/0.80	122.4
OF_2	45	1-1/8	0.05/0.80	114	198	1	95	0.05/0.80	165.3
CH_4	107	3/4	0.05/0.80	87	233	1	95	0.05/0.80	132.3
OF_2	-	-	—	-	178	1	60	0.05/0.80	143.2
B_2H_6	-	-	—	-	201	1/2	60	0.05/0.80	105.8
F_2	59	1/4	0.05/0.80	119	195	1-1/2	95	0.05/0.80	183.1
NH_3	16	1/4	0.3/0.95	68	198	1/4	95	0.3/0.95	85.5
N_2O_4	<15	1/4	0.6/0.91	86	165	1/4	65	0.6/0.91	133.8
A-50	<15	1/4	0.6/0.91	73	165	1/4	65	0.6/0.91	116.7
CIF_5	<15	1/4	0.6/0.91	75	165	1/4	65	0.6/0.91	113.4
MHF-5	<15	1/4	0.6/0.91	71	165	1/4	65	0.6/0.91	97.4

(a) P = tank pressure
(b) T = insulation thickness

In the idle mode, the engine starts with a tank pressure equal to that of the saturated propellant without aid from helium pressurization. After expulsion begins, helium is injected into the tank. The idle-mode start results in significant pressurant savings for the orbit trim burn because with this start mode it is unnecessary to precharge the ullage volume, which is 90 percent of total tank volume, with cold helium. (No heated helium is available until the engine is operating.)

The complete thermal-pressurization analysis was applied to the pressure-fed system studies, and included consideration of energy and mass transfer between ullage and liquid. The helium pressurant gas was heated before injection into the tanks. A parametric study was included to determine the optimum helium inlet temperature for each propellant.

All significant parameters that affected propellant tank size were considered in the analysis. These parameters included the helium storage sphere, which was assumed to be stored within the oxidizer or fuel tank, whichever had a lower temperature. The sphere temperature was assumed to be at the maximum liquid saturation temperature reached in the mission. The maximum helium storage pressure assumed was 4,500 psia.

The amount of pressurant required was computed using a modified Epstein Correlation to determine collapse factors and by conducting energy balances between the liquid and ullage. The propellant temperature response was computed taking into consideration external heat inputs and heating caused by injection of heated helium. Also considered were the effects of propellant vaporization into the ullage and liquid expansion caused by heating. Both heating and liquid expansion affect the initial ullage volume required. The initial propellant load was adjusted to account for vaporized liquid, which remains as vapor residuals. The change in tank size caused by all of these factors influences the external surface area and, therefore, heating through the insulation. All these parameters were considered parametrically, along with insulation thickness and total tank pressure, in order to compute a matrix of information from which all parameters are optimized.

All thermodynamic calculations were performed within the thermal-pressurization portion of the computer program, which includes correlations for temperature and pressure-dependent properties for all propellants.

The Summation and Plot routine was used to plot insulation, pressurant, pressurant sphere, vapor, and tank weights as a function of tank design pressure, with pressurant inlet temperature as a parameter. Also plotted is the sum of these weights, referred to as total system weight, as a function of tank design pressure. For example, the optimum tank pressure and helium inlet temperature for FLOX in a FLOX/CH_4 system is shown in Fig. 25. A typical plot of insulation, vapor, tank, and pressurant weights and their sum is shown in Fig. 26. Table 11 also shows optimum pressures and insulation thicknesses for the baseline Mars Orbiter pressure-fed system.

3.4 MARS ORBITER PROPULSION

The major effort in the area of propulsion was directed toward defining the engine systems in terms of their essential parameters, such as thrust, chamber pressure, mixture ratio, nozzle area ratio, envelope dimensions, weight, and other performance criteria. The support of the major engine companies was solicited to provide the data required for the analysis. The specific tasks performed are as follows:

- Definition of the essential engine parameters and requirements for each propellant formulation to be studied, including O_2/H_2, F_2/H_2, FLOX/CH_4, OF_2/CH_4, OF_2/B_2H_6, F_2/NH_3, N_2O_4/A-50, and ClF_5/MHF-5
- Resolution of design problems, such as secondary engine evaluation for mid-course correction and orbit trim, selection of cooling systems, and selection of nozzle designs for pump- and pressure-fed systems
- Integration of data and designs received from engine companies into finished engine parameters, and listing of the parameters for comparison and evaluation

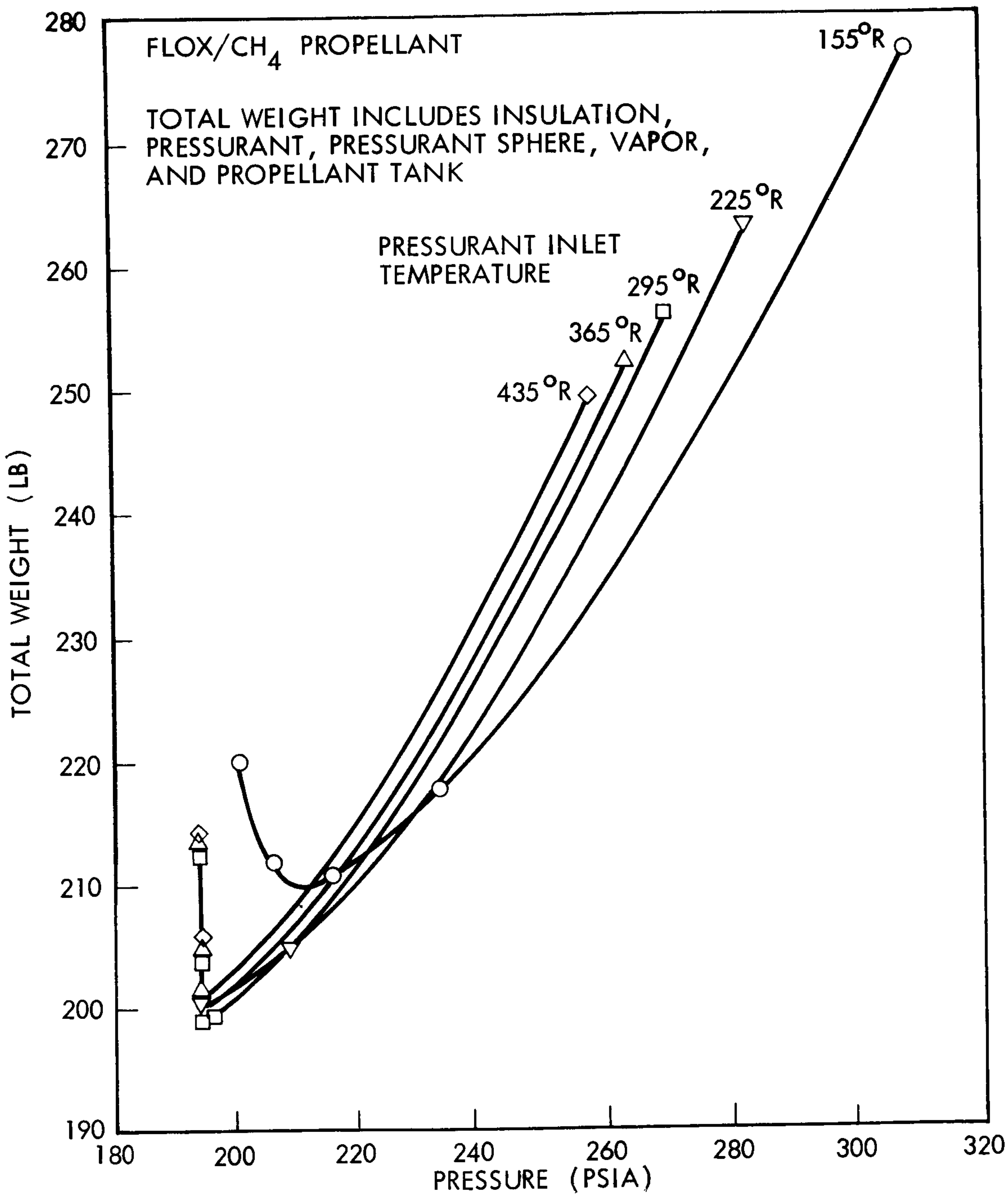


Fig. 25 Mars Orbiter – Pressure-Fed Optimization, FLOX/CH_4

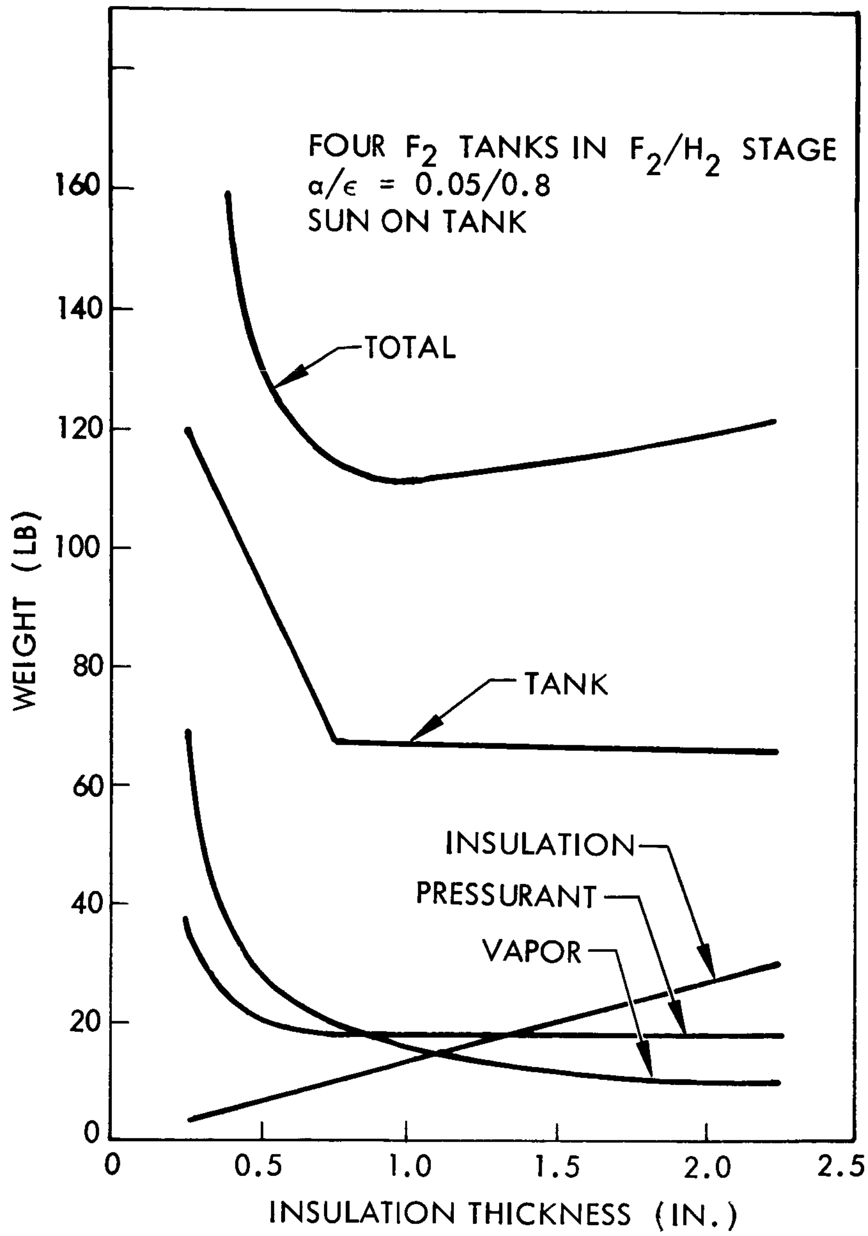


Fig. 26 Typical Pressure-Fed Mars Orbiter Optimization

3.4.1 Mars Orbiter Propellant Criteria

The propellant criteria considered significant by the engine companies included the following:

- Performance: I_{sp}, mixture ratio, recombination losses, etc.
- Storability: Temperature range
- Handling and safety
- Thermal stability
- Materials compatibility
- Ignition characteristics: Hypergolicity
- Cooling requirements
- Bulk density
- Cost

A detailed discussion of these criteria is presented in Volume III.

3.4.2 Mars Orbiter Engine Criteria

With the finalized propellant criteria, a second iteration was made with the engine companies in order to compile the specific engine data required for Task II. The engine companies were requested, as a minimum, to base their data on at least one point design for the 8,000 lbf (Mars Orbiter) engine, using both pump-fed and pressure-fed feed systems, and then to parameterize additional data for the various engine/propellant combinations.

The propulsion system performance parameters for the Mars Orbiter engine, as derived and refined from engine-company data, are listed in Table 12. This table reflects the nominal parameters that were employed for each engine/propellant combination, including propellant parameters, cooling type, nozzle shape, injector type, and engine size and weight.

Table 12

MARS ORBITER PROPULSION SYSTEM CHARACTERISTICS
(Bell Nozzle 8,000-lb Thrust)

Propellant	Mixture Ratio (O/F)	Pump-Fed Systems			Pressure-Fed Systems			Cooling
		Chamber Pressure (psia)	I_{sp} (a) (sec)	Engine Wt (lb)	Chamber Pressure (psia)	I_{sp} (a) (sec)	Engine Wt (lb)	
F_2/H_2	13	900	468	173 (b)	100	442	375	Regenerative
O_2/H_2	6	900	451	173 (b)	100	445	380	Regenerative
$FLOX/CH_2$	5.75	600	410	152	100	387	375	Regenerative
OF_2/CH_4	5.3	600	410	152	100	396	380	Regenerative
OF_2/B_2H_6	3.82	-	-	-	100	414	384	Ablative
F_2/NH_3	3.3	1,500	408	192	100	386	375	Regenerative
N_2O_4/A-50	2.0	750	335	158	100	328	330	Ablative
ClF_5/MHF-5	2.4	750	342	167	100	330	384	Ablative

(a) Based on $\epsilon = 100$
(b) Extendable bell nozzle

The nozzle shape selection was based on envelope requirements. The fixed bell shape was used wherever possible because of its light weight, high performance, and low cost. However, for the pump-fed O_2/H_2 and F_2/H_2 and all pressure-fed systems, the extendable nozzle shape was selected for the Mars Orbiter because of vehicle envelope limitations.

3.5 MARS ORBITER PROPULSION STAGE PERFORMANCE

Subsequent to the thermodynamic/structural optimization for each propellant combination, a complete vehicle was synthesized incorporating the structure, propellant feed assembly, pressurization system, engine system, contingency, residuals, and propellants. In addition, a complete vehicle was synthesized for each configuration with another propellant load so that the sensitivity to propellant load could be determined and scaling laws derived for the performance analysis.

The performance analysis was conducted with initial weight of the propulsion module as the performance figure of merit because the mission velocity and payload requirements are fixed.

For the pressure-fed design concepts, the total propulsion module weight is shown in Fig. 27. These are the baseline systems oriented with the sun on the tank for the interplanetary phase of the mission. The propulsion module with the lowest initial weight is the OF_2/B_2H_6 system, closely followed by the other space storables and the F_2/H_2 system. The high impulse-density of OF_2/B_2H_6 makes it a very promising propellant for a pressure-fed system. Table 13 lists the propulsion module weight, along with the fixed weights used as the basis in the TRW Voyager study in comparison with the current study results. The primary difference between the results obtained by TRW and those obtained in this study on N_2O_4/A-50 is due to the difference in specific impulse. A detailed weight breakdown of the propulsion module is shown in Table 14.

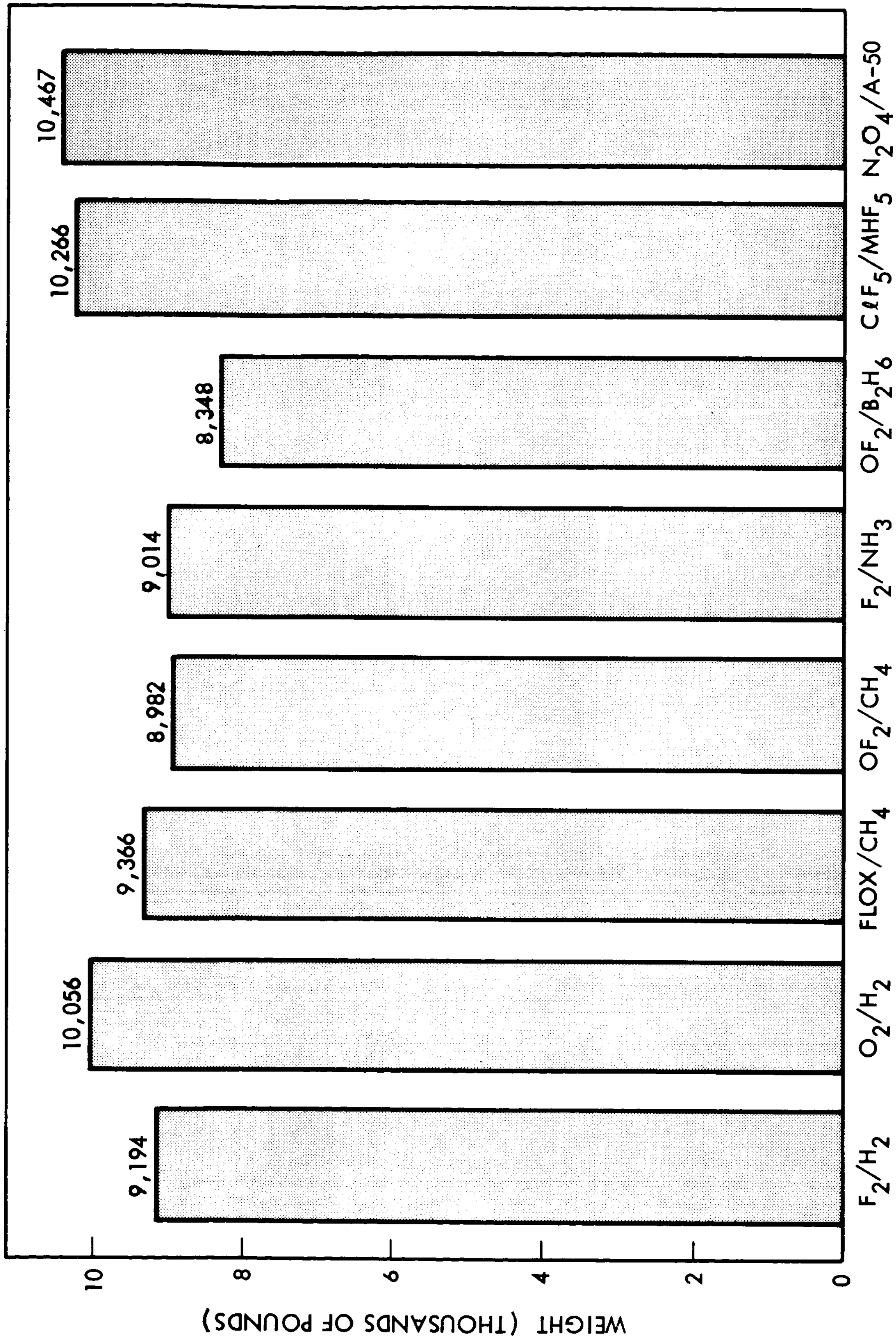


Fig. 27 Mars Orbiter Propulsion Module Weights – Pressure-Fed Systems
(Sun on Tanks, Nonvented, Optimum α/ϵ, 205-Day Mission)

Table 13

MARS ORBITER WEIGHT STATEMENT SUMMARY – PRESSURE-FED SYSTEM
(Sun on Tank, Nonvented, Optimum α/ϵ, 205-Day Mission)

ITEM	F_2/H_2	O_2/H_2	$FLOX/CH_4$	OF_2/CH_4	F_2/NH_3	OF_2/B_2H_6	ClF_5/MHF-5	N_2O_4/A-50	TRW BASELINE
CAPSULE	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000
SCIENCE SUBSYSTEM	400	400	400	400	400	400	400	400	400
CAPSULE SUPPORT	50	50	50	50	50	50	50	50	50
EQUIPMENT MODULE	1,980	1,980	1,980	1,980	1,980	1,980	1,980	1,980	1,980
PROPULSION MODULE									
INERT	2,596	3,164	1,997	1,893	1,782	1,757	1,565	1,633	2,382
PROPELLANT	6,598	6,892	7,369	7,089	7,232	6,591	8,701	8,834	11,071
VEHICLE WEIGHT MARGIN	713	713	713	713	713	713	713	713	713
TOTAL WEIGHT	17,337	18,199	17,509	17,125	17,157	16,491	18,409	18,616	21,596

Table 14

MARS ORBITER DETAILED WEIGHT BREAKDOWN – PRESSURE-FED SYSTEM
(Sun on Tank, Nonvented, Optimum α/ϵ, 205-Day Mission)

	WEIGHT (LB)							
	F_2/H_2	O_2/H_2	$FLOX/CH_4$	OF_2/CH_4	F_2/NH_3	OF_2/B_2H_6	ClF_5/MHF-5	N_2O_4/A-50
STRUCTURE								
BASE STRUCTURE	72	72	193	180	169	175	40	42
METEOROID PANELS	109	131	–	–	–	–	109	125
INTERNAL STRUCTURE	–	–	54	50	40	48	41	46
TANK SUPPORTS	90	96	119	112	102	108	96	104
ENGINE SUPPORT	96	96	–	–	–	–	–	–
ATTACHMENTS, ETC.	27	27	27	27	27	27	27	27
BULKHEAD INSULATION	45	45	45	45	45	45	45	45
PROPELLANT FEED ASSEMBLY								
TANKS	537	707	316	312	276	294	262	290
VALVES AND FILTERS	51	61	55	55	55	55	32	32
INSULATION	279	322	101	73	64	52	8	10
METEOROID BUMPER	117	129	98	93	86	87	-	-
PRESSURIZATION SYSTEM	176	304	139	115	101	105	119	162
ENGINE SYSTEM	375	380	375	380	375	384	384	330
CONTINGENCY	197	237	152	144	139	138	122	126
RESIDUALS								
PROPELLANT	86	106	91	87	93	82	133	139
VAPOR WEIGHT	146	210	89	76	67	49	3	3
He-GAS	86	160	34	38	24	19	17	16
PERFORMANCE RESERVE	107	81	109	106	119	89	127	138
PROPELLANTS	6,598	6,892	7,369	7,089	7,232	6,591	8,701	8,834
PROPULSION MODULE WEIGHT	9,194	10,056	9,366	8,982	9,014	8,348	10,266	10,467

The pump-fed propulsion module weights for the baseline system are shown in Fig. 28. For these combinations, the F_2/H_2 propellants provide the lightest weight system. A detailed weight breakdown is shown in Table 15. A comparison among typical cryogenic, space storable, and earth storable pump-fed and pressure-fed systems is shown in Table 16. For the conditions assumed, i.e., constant engine thrust and expansion ratio, the engine and pressurization systems are uniformly heavier for the pressure-fed system. For the earth storables, the propellant feed system is approximately the same weight for both the pump- and pressure-fed systems. For the space storables the propellant feed system is somewhat heavier for the pressure-fed system, and for the cryogens the larger inert weight and the storage of pressurant in the H_2 tank had a significant impact on the propellant feed system weight. The ranking of the initial weights of all the pressure-fed and pump-fed systems is given in Table 17. As shown in the table, the F_2/H_2 system is the lightest pump-fed system and the lightest of all systems. The OF_2/B_2H_6 system is the lightest pressure-fed system, and ranks fifth overall in a group of 15 candidates.

3.6 SUMMARY OF MARS ORBITER ANALYSIS

The Mars Orbiter stage provided a very good vehicle for a propellant selection comparison. It consisted of a system that had a significant mission duration, was of intermediate size, and provided the possibility for many sensitivity analyses.

A minimum of design modification was required for this system, although the hydrogen propellant cases required ellipsoidal tanks. Both pump and pressure-fed systems were considered. This not only required different thermodynamic and pressurization analyses but also yielded significantly different thermodynamic/structural optimization results. The engine weights and specific impulse values also were very different between the pump- and pressure-fed systems. This resulted in performance characteristics which indicated that the F_2/H_2 propellants provided the lightest weight pump-fed system and the OF_2/B_2H_6 propellants provided the lightest pressure-fed system.

Results of Mars Orbiter Sensitivity analyses are presented in Section 6.

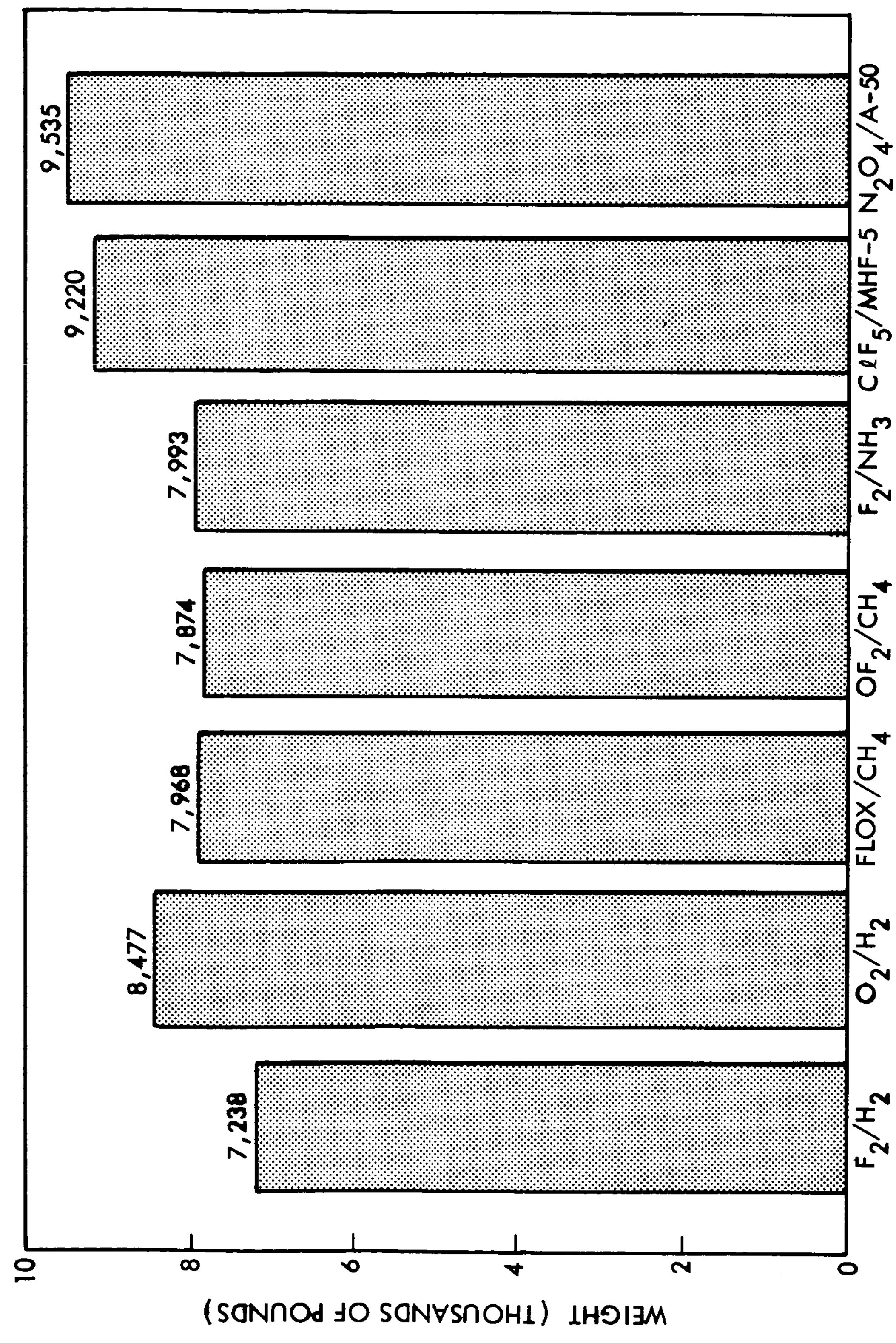


Fig. 28 Mars Orbiter Propulsion Module Weights — Pump-Fed Systems (Sun on Tanks, Nonvented, Optimum α/ϵ, 205-Day Mission)

Table 15

MARS ORBITER DETAILED WEIGHT BREAKDOWN – ~~PRESSURE~~ PUMP-FED SYSTEM
(Sun on Tank, Nonvented, Optimum α/ϵ, 205-Day Mission)

	WEIGHT (LB)						
	F_2/H_2	O_2/H_2	$FLOX/CH_4$	OF_2/CH_4	F_2/NH_3	ClF_5/MHF-5	N_2O_4/A-50
STRUCTURE							
BASE STRUCTURE	72	72	178	170	170	33	36
METEOROID PANELS	78	122	–	–	–	119	120
INTERNAL STRUCTURE	–	–	50	48	49	40	41
TANK SUPPORTS	69	94	110	108	106	96	101
ENGINE SUPPORT	91	91	–	–	–	–	–
ATTACHMENTS, ETC.	27	27	27	27	27	27	27
BULKHEAD INSULATION	45	45	45	45	45	45	45
PROPELLANT FEED ASSEMBLY							
TANKS	311	497	290	287	287	275	301
VALVES AND FILTERS	51	61	55	55	54	33	32
INSULATION	233	372	73	61	61	15	18
METEOROID BUMPER	71	140	75	74	72	–	–
PRESSURIZATION SYSTEM	36	40	59	54	45	28	24
ENGINE SYSTEM	152	152	152	152	192	167	158
CONTINGENCY	125	172	114	109	111	87	90
RESIDUALS							
PROPELLANT	86	106	91	87	93	133	139
VAPOR WEIGHT	126	179	77	64	64	4	4
He-GAS	5	7	11	12	7	2	2
PERFORMANCE RESERVE	73	80	76	75	92	122	137
PROPELLANTS	5,587	6,220	6,485	6,446	6,518	7,994	8,260
PROPULSION MODULE WEIGHT	7,238	8,477	7,968	7,874	7,993	9,220	9,535

Table 16

MARS ORBITER – PRESSURE-FED/PUMP-FED WEIGHT COMPARISON
(Sun on Tanks, Nonvented, Optimum α/ϵ, 205-Day Mission)

PROPULSION MODULE ELEMENT	F_2/H_2		OF_2/CH_4		N_2O_4/A-50	
	PUMP-FED (LB)	PRESS.-FED (LB)	PUMP-FED (LB)	PRESS.-FED (LB)	PUMP-FED (LB)	PRESS.-FED (LB)
STRUCTURE	382	439	398	414	370	389
PROPELLANT FEED ASSEMBLY	666	984	477	533	351	332
PRESSURE SYSTEM	36	176	54	115	24	162
ENGINE SYSTEM	152	375	152	380	158	330
CONTINGENCY	125	197	109	144	90	124
RESIDUALS	217	318	163	201	145	158
PERFORMANCE RESERVE	73	107	75	106	137	138
PROPELLANT	5,587	6,598	6,446	7,089	8,260	8,834
TOTAL MODULE WEIGHT (LB)	7,238	9,194	7,874	8,982	9,535	10,467

Table 17

MARS ORBITER STAGE WEIGHT
(Sun on Tanks, Nonvented, Optimum α/ϵ, 205-Day Mission)

RANK	PUMP-FED	PRESSURE-FED	WEIGHT (LB)
1	F_2/H_2		7,238
2	OF_2/CH_4		7,874
3	$FLOX/CH_4$		7,968
4	F_2/NH_3		7,993
5		OF_2/B_2H_6	8,348
6	O_2/H_2		8,477
7		OF_2/CH_4	8,982
8		F_2/NH_3	9,014
9		F_2/H_2	9,194
10	ClF_5/MHF-5		9,220
11		$FLOX/CH_4$	9,366
12	N_2O_4/A-50		9,535
13		O_2/H_2	10,056
14		ClF_5/MHF-5	10,266
15		N_2O_4/A-50	10,467

Section 4
MARS EXCURSION MODULE ASCENT-STAGE INVESTIGATION

4.1 MEM INPUTS AND ASSUMPTIONS

The Mars Excursion Module (MEM) Ascent Stage was one of two vehicles selected by NASA for further investigation. The selected concept was the Apollo-shaped MEM as defined by North American-Rockwell Co. and shown in Fig. 29. A detailed description is presented in Appendix C. The selected mission designation consisted of the following:

- 1982 Mars Lander
- 221-Day Mission Duration
  - 30 Days Earth Orbit
  - 161 Days Interplanetary Cost
  - 30 Days Mars Surface
- 270 Nautical Mile Circular Mars Orbit

To assess the vehicle performance and conduct the ascent vehicle analysis a 5,260-lb ascent stage capsule weight was assumed. Other vehicle requirements were as follows:

- Four-man/30-day vehicle
- Ascent ΔV = 16,000 ft/sec (13,800 ft/sec first burn and 2,200 ft/sec second burn)
- Ascent thrust = 30,000 lb

Only pump-fed systems were investigated as the primary candidates.

The propellants used in this analysis include the following:

- F_2/H_2
- O_2/H_2 and O_2/H_2 subcooled

- FLOX/CH_4
- OF_2/CH_4
- OF_2/B_2H_6 (limited analysis using a pressure-fed system)
- F_2/NH_3
- ClF_5/MHF-5

4.2 MEM ASCENT STAGE DESIGN

4.2.1 MEM Ascent Stage Design Assumptions and Ground Rules

The design effort on the MEM ascent stage has included (1) securing reference North American-Rockwell (NAR) Space Division baseline configuration data, (2) formulating a design approach, (3) preparing preliminary ascent stage propellant tankage layouts for the various propellant combinations involved, and (4) synthesizing complete ascent stages.

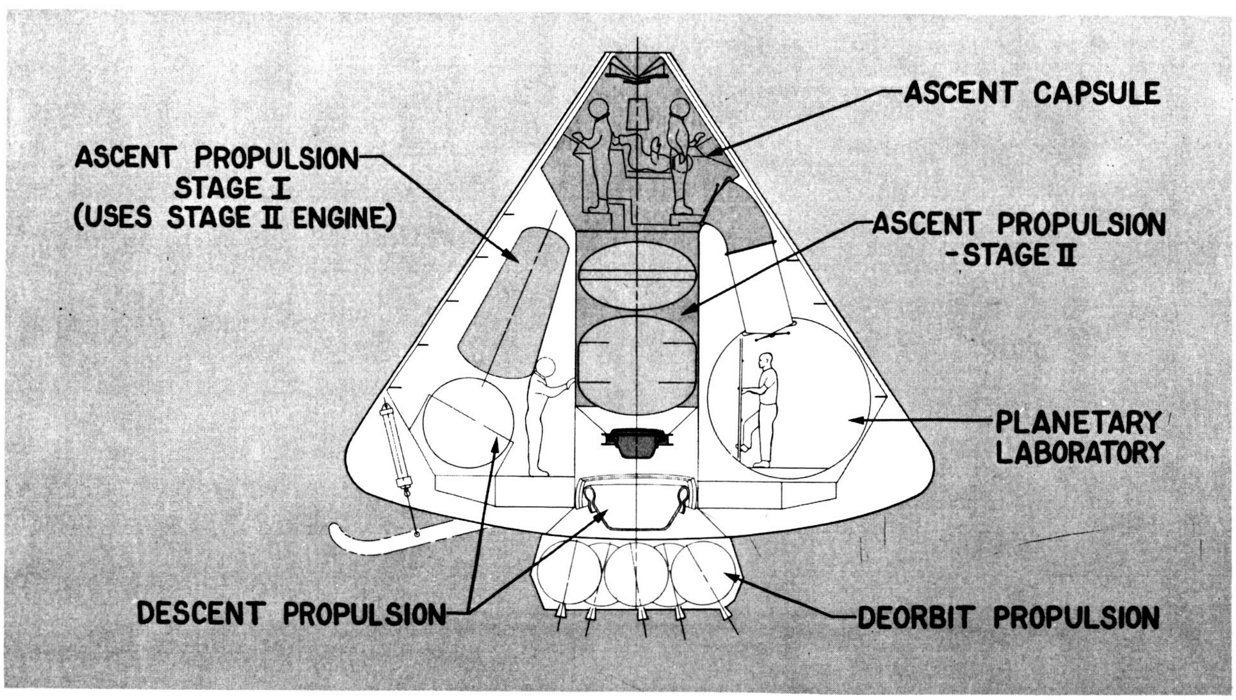


Fig. 29 Baseline MEM – North American-Rockwell Co.

The MEM, as defined by NAR is an Apollo-shaped vehicle, and is one unit of the total Earth-Mars Aerobraker spacecraft. On an Earth-Mars mission, the Aerobraker spacecraft will enter a Mars orbit. The MEM vehicle is then separated from the Aerobraker and descends to the surface of Mars. After a stay of 30 days, the ascent stage of the MEM vehicle will lift off the surface of MARS and rendezvous with the orbiting Aerobraker. The LMSC design task is centered on this MEM ascent stage.

The NAR MEM ascent stage design data were reviewed in detail by LMSC. The ascent stage was divided into Stages I and II. Stage I includes a set of droppable propellant tanks and their attached structure, and Stage II includes the ascent stage, main body, ascent engine, and another set of propellant tanks. The Stage I and Stage II designations were established in the design area for the purpose of clarity, and does not imply a performance requirement, since optimum Mars ascent trajectory studies have not been a part of the design discussion that follows.

From this analysis of the NAR MEM ascent stage data, the following LMSC design approach ground rules were established:

(1) MEM lander diameter was held to a constant 30-ft diameter. 31.5 ft was used for O_2/H_2, which is the upper limit to permit enclosure within the 33-ft-diameter Aerobraker.

(2) NAR – defined MEM descent stage, shape, and volume, including FLOX/CH_4 tankage and laboratory, were held constant.

(3) NAR ascent stage main body structure and crew capsule size were held constant.

(4) NAR – defined conical propellant tankage was revised on the ascent Stage I to spherical, except for the O_2/H_2 propellant case, which still required conical tankage for the hydrogen propellant (the O_2 propellant can be spherical) to allow any hope of packaging within available space on the 31.5-ft-diameter lander.

(5) The NAR-defined 2:1 ratio elliptical tank bulkhead was revised in all cases to a $\sqrt{2}$:1 ratio.

(6) Stage II eliptical propellant tankage was held to a 70-in. (or less) diameter to fit within the 76-in.-diameter main body shell.

(7) All ascent Stage I and II propellant tanks are attached and supported with trusses composed of low-heat-leak struts. These struts are arranged to produce the minimum number of struts possible per tank (and thus minimum heat leak).

(8) Flight load factors include:

	Axial (g_0)	Lateral (g_0)
• Earth Departure – Max αq	2	3
– Max	5	0.5
• Mars Capture	-10	±3
• Mars Landing	+5	±2

(9) Assumed nominal ullage volumes were 3 percent for earth storables, 7 percent for H_2, and 5 percent for all other propellants.

(10) Meteoroids:

- Flux and penetration models from Refs. 1 and 2 (modified)
- Probability of no penetration = 0.99
- Tank exposure = 30 days on surface of Mars
- Propellant tanks are not to be used as part of the shield

(11) Thermal ground rules include earth surface to separation from Aerobraker at Mars, fully enclosed in Aerobraker. MEM stowage compartment environment to be determined. Assuming rotation at 4 rpm in plane of ecliptic and Aerobraker skin $\alpha/\epsilon = 0.25$.

(12) Mars atmosphere – Model VM-7

(13) Design factor of safety is 1.40 to ultimate stress at margin of safety = zero. Check for no yield at limit, which is 1.1 times maximum applied load.

(14) Materials

- Tanks – Use welded 2021 aluminum for all propellant tanks where suitable
- Tank Supports – Low-heat-leak supports as appropriate, based on LMSC experience (probably fiberglass).
- Insulation – Type to be determined by environmental effects (evacuated insulation probably required while on the surface of Mars)

Preliminary LMSC packaging/performance studies of various lander diameters (other than the NAR-defined 30-ft diameter) indicated that only nominal advantages were to be gained by widely varying the lander diameter. This is due largely to the MEM physical envelope and descent stage constraints, which tend to reduce the propellant packaging flexibility that is normally obtained in varying physical size.

The primary objectives of the MEM ascent stage design study based on the established design approach discussed were as follows:

- Establish realisitc design data for thermodynamic propellant storage studies
- Establish parametric weight data for all structural and other related elements of the ascent vehicle, for both Stages I and II

4.2.2 MEM Ascent Stage Design Analysis

Before preliminary design layouts of the MEM ascent vehicle could begin, it was necessary to know the propellant loading requirements for each propellant combination. To define these preliminary propellant loadings, a preliminary performance analysis was made for all propellant combinations. Using these preliminary propellant loadings, it was then possible to complete a series of ascent vehicle layouts that defined the following.

- Propellant tankage configurations and arrangements for stages I and II of the ascent vehicle for all propellant combinations
- Tank support truss arrangements for follow-on structural and thermal analysis inputs

A general arrangement layout of a typical ascent vehicle is shown in Fig. 30. The tables on the drawing show 1st and 2nd stage tank configurations, tank size, the number of tanks required for each propellant combination, engine data, etc. The ascent stage general arrangement indicates detail of tankage attachment for all storable configurations. For the O_2 propellant combination, detailed scale layouts were completed of a MEM configuration with a base diameter of 31.5 ft. On this configuration, the propellant tanks were made conical to obtain maximum H_2 propellant packaging efficiency within the geometric and physical constraints of the 31.5-ft-diameter vehicle. However, the O_2/H_2 combination would not meet the MEM volume constraints.

Detailed scale layouts were also drawn for the other tank configurations to define the volumetric and truss arrangements necessary to establish the individual tube lengths and angles with respect to the applied loads for the strut loads analysis.

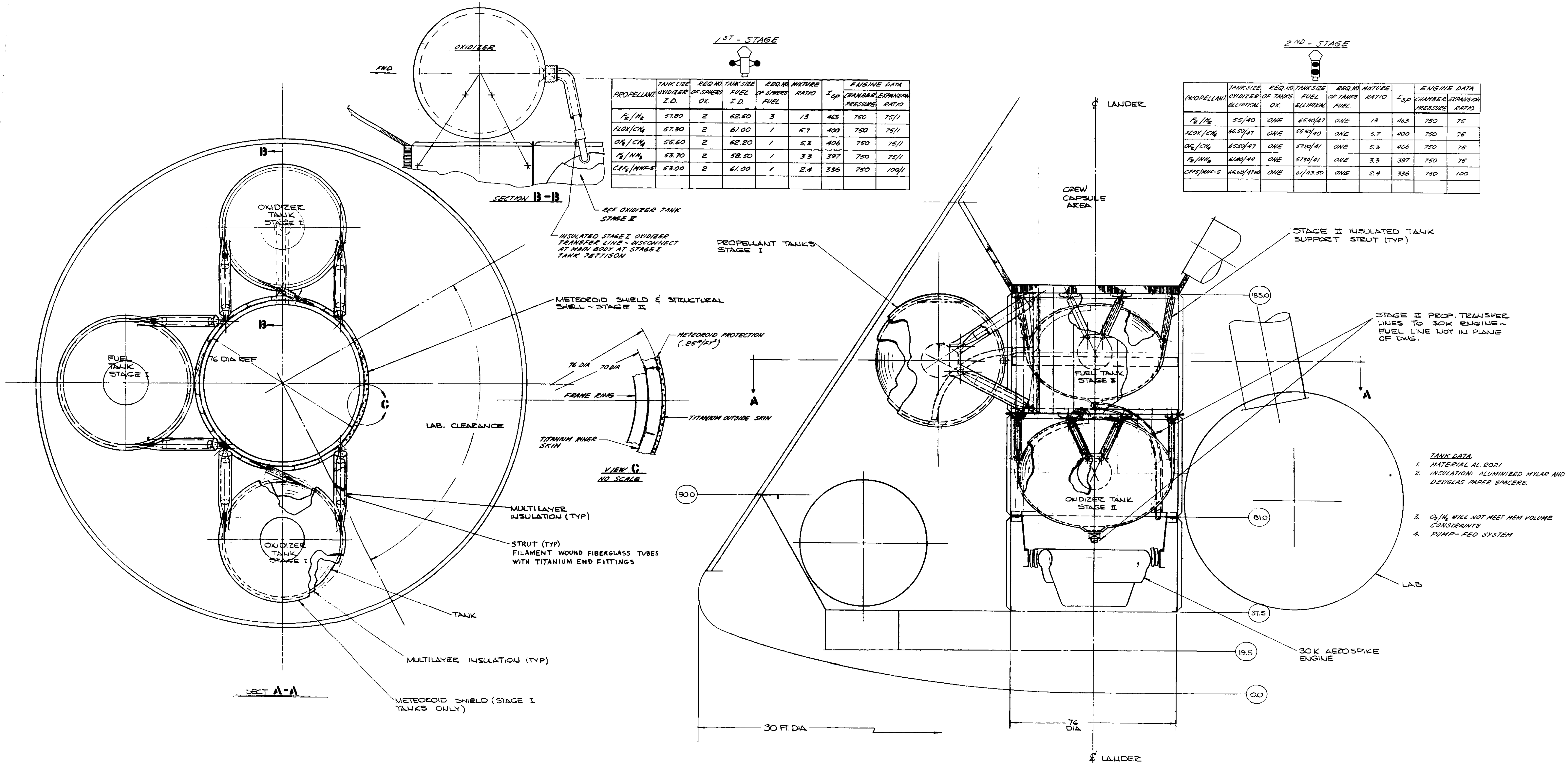
An item of special interest is the propellant packaging efficiency and growth potential possible in each configuration over the basic ΔV design point of 16,000 ft/sec. General conclusions on growth potential for the various propellant combinations are as follows:

- All propellant combinations except the F_2/H_2 systems have growth potential in the Stage I portion of the ascent vehicle. This potential takes the form of space available for more propellant packaging.
- Hydrogen systems would require increases in lander diameter coupled with modifications to tankage arrangements and shape to effect significant performance growth capabilities.

4.2.3 MEM Ascent Stage Structural Analysis

The LMSC MEM ascent vehicle structural analysis conducted included obtaining parametric weight data versus propellant combination and weight variations for the following main structural elements of the ascent vehicle:

- Tank parametric weight analysis for a range of limit pressures to 300 psi
- Tank attach and support truss parametric weight analysis using low-heat-leak fiberglass struts
- 76-in.-diameter monocoque ascent stage shell loads, stress, and parametric weight analysis



1ST - STAGE

PROPELLANT	TANK SIZE OXIDIZER I.D.	REQ NO. OF SPHERS OX.	TANK SIZE FUEL I.D.	REQ. NO. OF SPHERS FUEL	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO
F_2/H_2	57.80	2	62.50	3	13	463	750	75/1
$FLOX/CH_4$	57.30	2	61.00	1	5.7	400	750	75/1
OF_2/CH_4	55.60	2	62.20	1	5.3	406	750	75/1
F_2/NH_3	53.70	2	58.50	1	3.3	397	750	75/1
$ClF_5/MHF\text{-}5$	53.00	2	61.00	1	2.4	336	750	100/1

2ND - STAGE

PROPELLANT	TANK SIZE OXIDIZER ELLIPTICAL	REQ. NO. OF TANKS OX.	TANK SIZE FUEL ELLIPTICAL	REQ. NO. OF TANKS FUEL.	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO
F_2/H_2	55/40	ONE	65.40/47	ONE	13	463	750	75
$FLOX/CH_4$	66.50/47	ONE	55.50/40	ONE	5.7	400	750	75
OF_2/CH_4	65.50/47	ONE	57.20/41	ONE	5.3	406	750	75
F_2/NH_3	61.80/44	ONE	57.30/41	ONE	3.3	397	750	75
$ClF_5/MHF\text{-}5$	66.50/47.50	ONE	61/43.50	ONE	2.4	336	750	100

Fig. 30 MEM Typical Ascent Stage Design

Tanks. The spherical and elliptical aluminum tank parametric weight data developed for the Mars orbiter were used for the MEM without modification because the range of limit pressures, tank sizes, and design criteria were compatible between the two systems. The tanks for the MEM are of a somewhat different design, however, because they operate in a tangible atmosphere on the Martian surface. Figure 31 shows a concept for a dual-wall, space-evacuated tank system assumed for the analysis. The duel-wall space contains insulation and provides the second wall of the meteoroid shield.

Geometry of Truss Members. The Stage I and Stage II propellant tanks of the MEM ascent vehicle are attached and supported by tubular truss configurations as shown in Fig. 32. The primary design criteria of these truss arrangements are to (1) produce the minimum possible heat leak between the propellant tank and the 76-in.-diameter main body monocoque shell support structure and (2) to distribute the applied loads in such a manner as to reduce the individual truss member loads and reacting attach point loads to a minimum.

Truss Loads Analysis. A truss loads analysis was completed using the previously defined truss configurations and geometric data. The objective of this loads analysis was to establish maximum strut loads in all truss arrangements for all propellant combinations.

Loading conditions were calculated for each flight condition for each truss member and all propellant combinations. From this matrix of loads, worst-case values were defined and combined in the analysis procedure discussed in the following paragraphs.

The first step in this analysis was to establish the force systems for each truss configuration. The second step was to calculate the individual main strut net loads. The loads were based on the following applied load assumptions:

- Propellant tank center of gravity was located at the geometric center of the tank assembly.

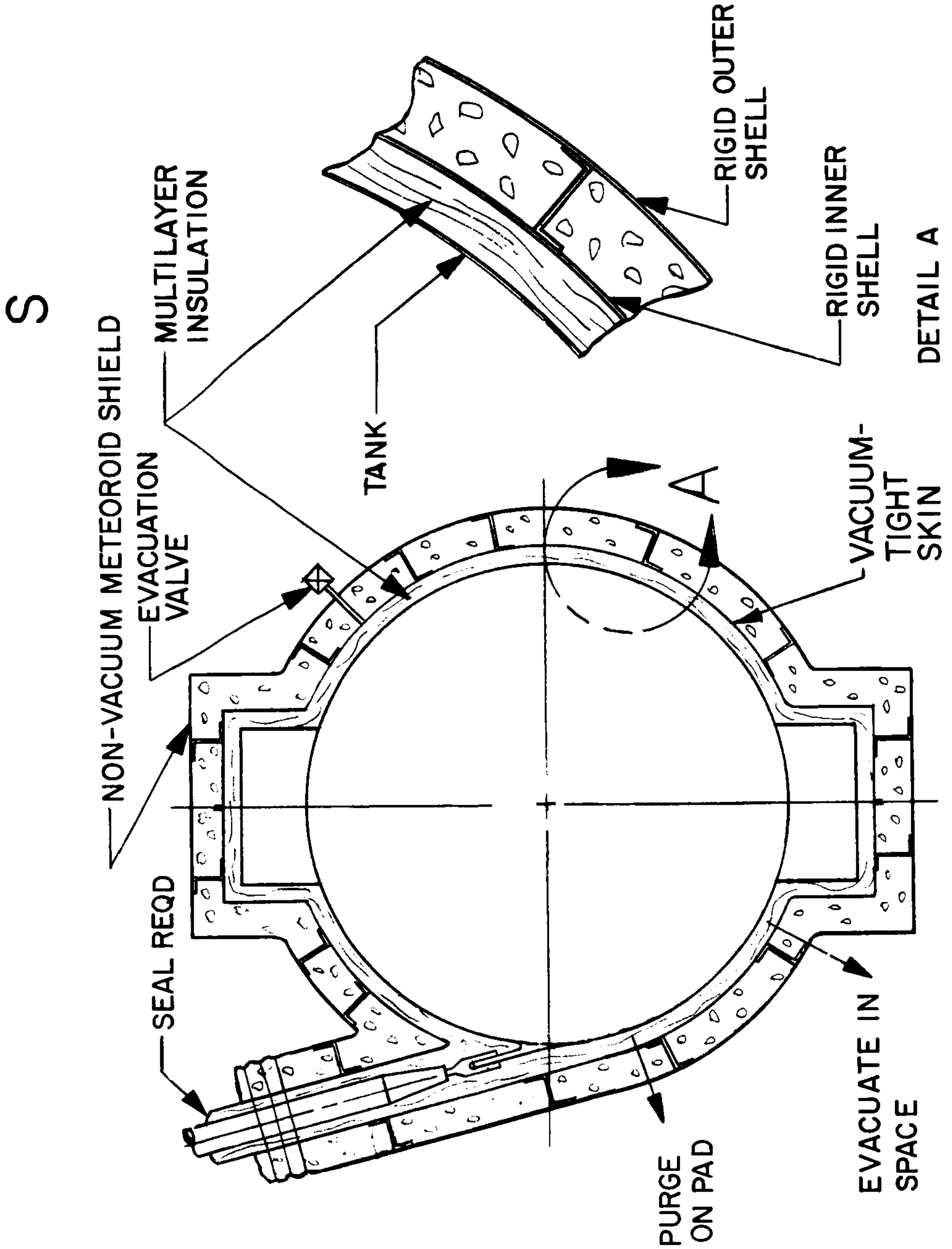


Fig. 31 MEM Tank Design Details

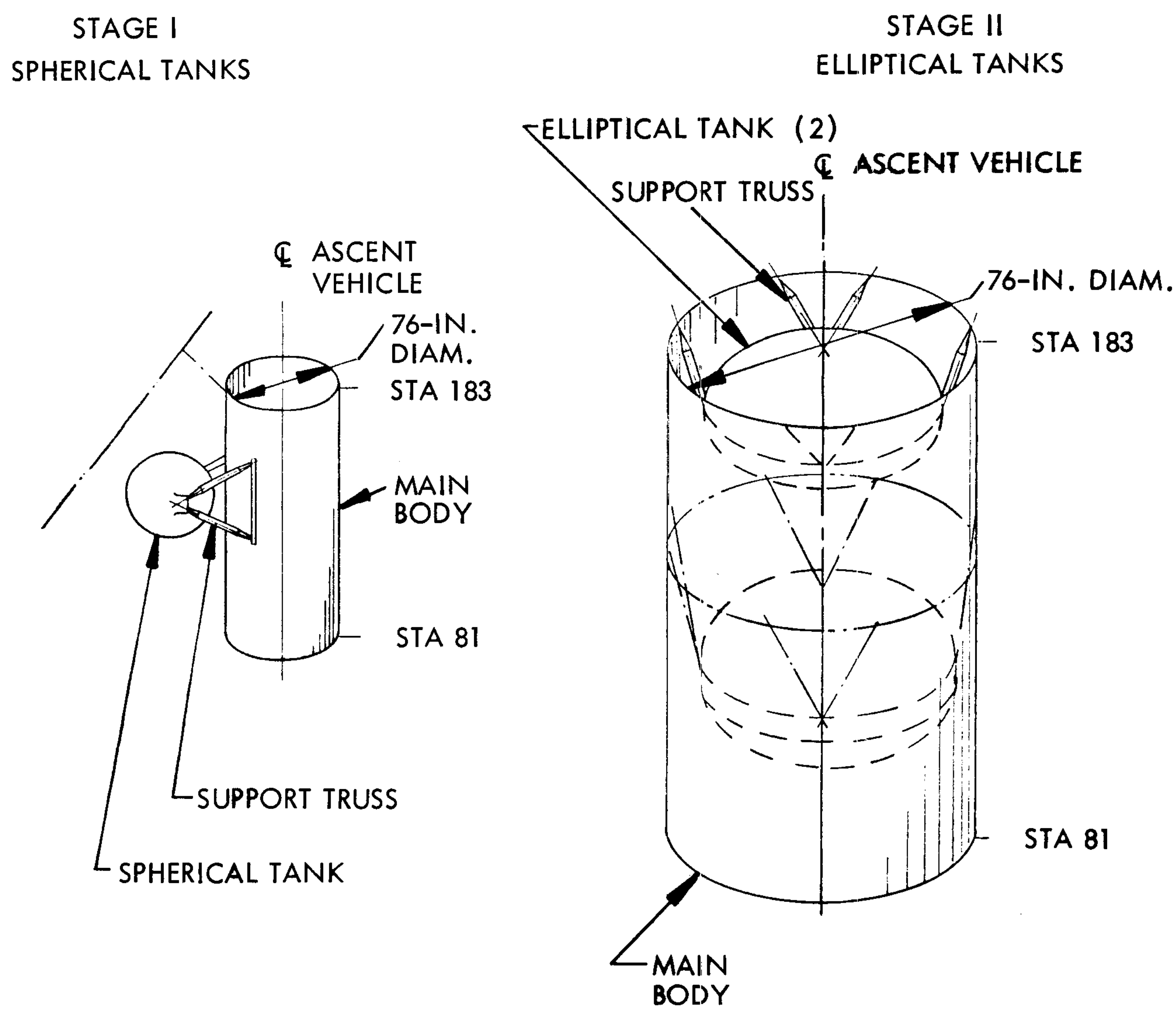


Fig. 32 MEM Tank Support Arrangement

- Applied loads caused by flight accelerations were assumed to act normal to or parallel with the ascent vehicle geometric longitudinal center line.
- Lateral loads were assumed to act in a bending sense only on the truss assemblies. This was assumed to be the worst case.

Parametric Weight Analysis of Truss Members. A parametric weight analysis of all truss members was completed for Stages I and II of the ascent stage. A brief weight and thermal comparison of an all-titanium strut versus the fiberglass/titanium end fitting strut design was also completed to establish design comparison data. For equal design conditions, results indicated a weight savings of approximately 30 percent could be realized for an all-titanium strut versus a fiberglass strut; however, the strut heat leak would increase by a factor of 10 to 12. On this basis, it was decided to use fiberglass struts for truss members in the detailed weight analysis. In determining total truss weights, the following was applied for the tank truss: Truss weight = (sum of all strut weights) × 1.05, where 1.05 is a 5 percent contingency factor.

Ascent Stage Shell Analysis. The main 76-in.-diameter, 102-in.-long body of the ascent vehicle extends from the bottom of the crew capsule to the thrust structure of the 30,000-lb-thrust ascent engine. This results in ascent vehicle structural integrity and provides a means for attaching and supporting the Stage I and II ascent propellant tankage. It also provides the primary physical attach and separation interface with the descent portion of the MEM vehicle, as well as many other miscellaneous attach/support points for various items of equipment, wiring, plumbing, etc.

The main body is configured as a skin, stringer, frame-type structure. In addition to carrying basic running load distributions from various sources, it must also pick up and support a series of concentrated loads introduced by the propellant tank support struts. This second requirement introduces the need for a series of relatively heavy, tapered longerons both inside and outside the shell, which would be integrated with reinforced frames. This structural arrangement is shown conceptually in Fig. 33.

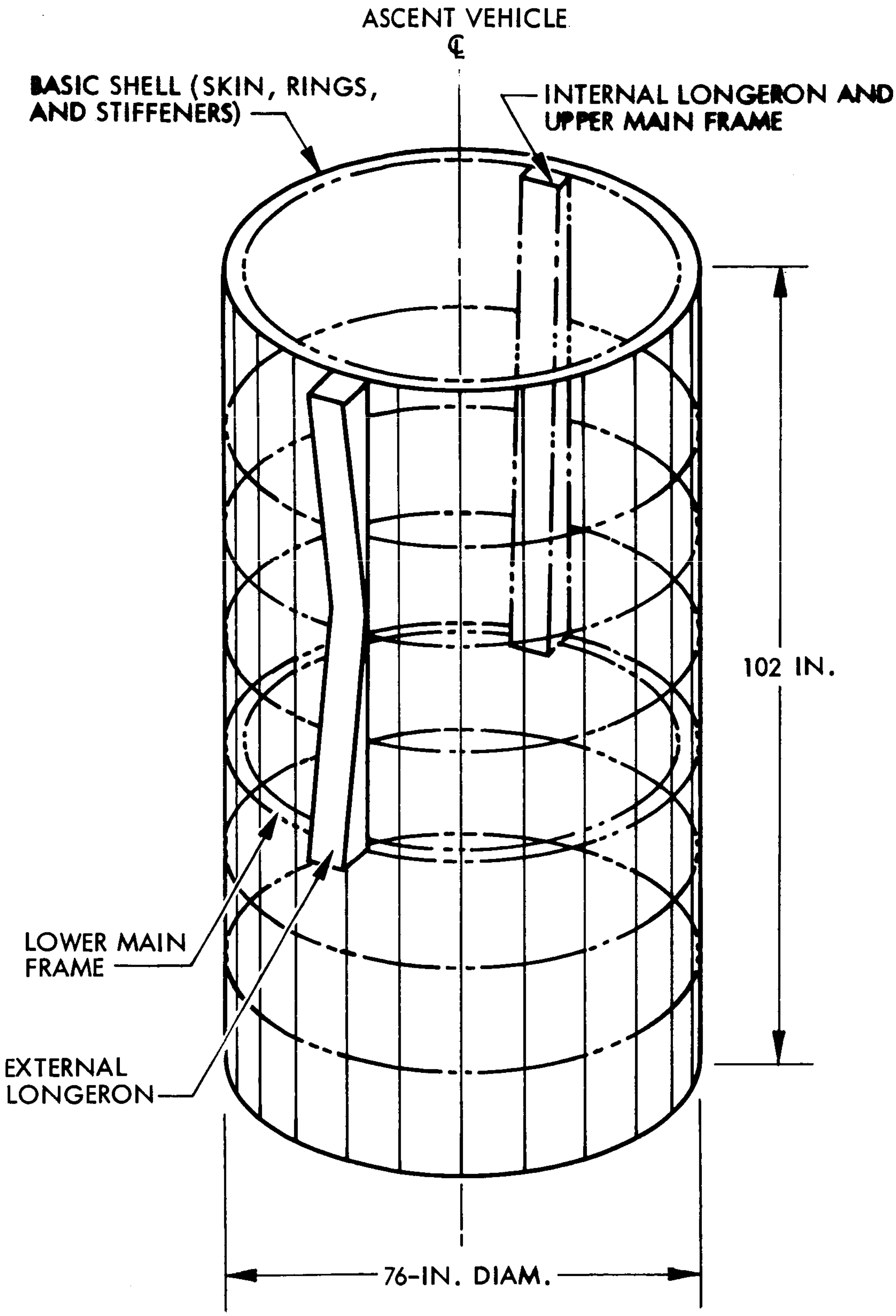


Fig. 33 Ascent Stage Main Body Shell Concept

The LMSC analysis was based on the following approach:

- Maximum running load distributions were determined for worst loading conditions. Semi-monocoque shell weights due to these running loads were then determined using an off-the-shelf LMSC computer program.
- Concentrated loads were then introduced into this basic shell and distributed to allowable running load or shear limits (through the use of tapered longerons and frames).
- The weight ranges of these added-on longerons and frames were parametrically determined and added to the basic shell weights defined by the computer program, along with suitable contingency factors. Titanium material was assumed for this structure because of its superior strength-to-weight and stiffness-ratio properties.

4.3 MEM ASCENT STAGE THERMODYNAMIC AND PRESSURIZATION ANALYSIS

4.3.1 Thermodynamic Analysis Procedures

Analyses for the MEM were based on the same type of approach (Fig. 21) as for the Mars Orbiter. Only pump-fed systems were analyzed because of the volumetric considerations of the MEM vehicle.

A thermal model representing the MEM was developed with fixed inner tank (Stage 2). The number and shape of external tanks were adjusted in the model for each propellant combination. Assumptions applicable to the MEM are as follows:

- NPSP = 4 psia
- Heated helium pressurization system except hydrogen bleed for H_2 tanks
- Inner and outer tanks interconnected
- Ullage accommodated in outer tanks
- Engine heat-soak back eliminated by postflow
- Engine preconditioning accomplished by preflow
- Nonvented tanks

4.3.2 Mars Excursion Module System Analysis – Thermodynamics and Pressurization

The MEM mission included three major phases: (1) 30 days in earth orbit at an altitude of 270 nm with an orbit plane-solar incidence angle β of 52 deg, (2) a 160-day Earth-Mars transit with the vehicle tumbling end over end in a plane containing the solar vector, and (3) a 30-day stay period on the Mars surface. During earth orbit and transit phases, the MEM is enclosed within the Aerobraker. Heating of the MEM in earth orbit caused by both earth emission and albedo was very significant and was accounted for in the analysis. During the Mars surface stay, the MEM ascent stage is exposed directly to the environment. Propellant heating during the Mars entry phase was assumed negligible because the heat shield absorbs the aerodynamic heating load and is ejected. Heating of the propellant during the short (up to 8 hr) period between first and second burns of the ascent stage is accounted for in the analysis, although the effect is slight.

While on the Mars surface, the MEM is assumed located on the equator, and is constantly subjected to convective heating resulting from the density-velocity specified by the JPL VM-7 Mars atmosphere. Preliminary studies showed that the pressure levels in the Mars atmosphere would degrade the performance of multilayer insulation by at least a factor of 200 relative to evacuated performance. It was then established that an evacuated enclosure would have to be provided for insulation on cryogenic and space-storable propellant tanks. Evacuation of insulation for earth-storable propellants is not required.

The MEM configuration had fixed inner tanks (Stage 2). The outer tanks were assumed to be spherical except for the H_2/O_2 system, which used conical hydrogen tanks. The size and number of outer tanks varied for each propellant combination. Because all of the propellant in the outer tanks and some of that in the inner tanks is used for the first burn, the outer and inner tanks are plumbed in series. That is, the propellant from the outer tank is fed through the inner tank. Valves in the connecting lines are never closed until after Stage 1 shutdown, at which time the interconnecting lines are closed by squib valves before the outer tanks are dropped. Because of this connection, inner and outer tanks always experience the same pressure until the valves are closed at staging. The inner tanks can then be pressurized to a different level for the final burn.

Considerable liquid expansion occurred for all propellants, particularly for H_2 and CH_4. Initial ullage requirements include an allowance for liquid expansion from the full inner tanks into the outer tanks.

Results of the MEM optimization study are presented in Table 18, where optimum insulation thicknesses and tank design pressures are given for the inner and outer tank. Ullage volume requirements are also shown. The ullage is contained totally in the outer tanks. The outer tank pressure never exceeds inner tank pressure because of the interconnection. However, the inner tank pressure, if optimum, was allowed to exceed the outer tank pressure after first-stage burn.

4.4 MEM ASCENT STAGE PROPULSION

The propulsion system selection for the MEM was developed in a manner similar to that for the Mars Orbiter. This involved defining the engine systems used for each selected propellant combination in terms of their essential parameters, such as thrust, chamber pressure, mixture ratio, nozzle area ratio, envelope dimensions, weight, and other performance criteria. The following specific tasks were performed:

- Definition of the essential engine parameters and requirements for each propellant formulation to be studied, including O_2/H_2, F_2/H_2, FLOX/CH_4, OF_2/CH_4, F_2/NH_3, and $C\ell F_5$/MHF-5
- Resolution of design problems and selection of combustor and nozzle cooling systems
- Integration of data and designs received from engine companies into finished engine parameters, and listing of the parameters for comparison and evaluation

Table 18

MEM THERMODYNAMIC OPTIMIZATION

Propellant	α/ϵ Tank	α/ϵ Aero-Braker	Total Tank Pressure (psia)		Insulation Thickness (in.)		Initial Ullage (%)
			Outer	Inner	Outer	Inner	
O_2	0.05/0.8	0.05/0.8	61	61	1/2	1	10
H_2	0.05/0.8	0.05/0.8	79	79	4	4	16
F_2	0.05/0.8	0.05/0.8	46	60	3/4	1	9
H_2	0.05/0.8	0.05/0.8	102	120	4	3 1/2	23
FLOX	0.05/0.8	0.05/0.8	56	67	3/4	1	8
CH_4	0.05/0.8	0.05/0.8	50	54	1/2	3/4	17
OF_2	0.05/0.8	0.05/0.8	37	54	1/2	1/2	9
CH_4	0.05/0.8	0.05/0.8	49	53	1/2	3/4	12
F_2	0.05/0.8	0.05/0.8	52	70	7/8	1	9
NH_3	0.2/0.9	0.05/0.8	42	64	1/2	1/2	5
ClF_5	0.93/0.88	0.93/0.88	27	34	min	min	3
MHF-5	0.93/0.88	0.93/0.88	<15	<15	min	min	3

4.4.1 MEM Ascent Stage Propellant Criteria

The propellant evaluation effort paralleled that performed for the Mars Orbiter, as previously described.

4.4.2 MEM Ascent Stage Engine Criteria

Additional parameters and resolutions required for the MEM ascent engine were as follows:

- Selection of engine type was dictated by required dimensional constraints of the overall vehicle envelope. The type selected was the Aerospike engine, as proposed by Rocketdyne. A comparison of engine diameter and length for the pump-fed versions of the Bell, Extended Bell, and Aerospike designs for the MEM ascent engine (at $\epsilon = 100$, using $FLOX/CH_4$ propellant) is as follows:

Type	Diameter, (in.)	Length, (in.)
Bell	64	140
Extended Bell	64	64
Aerospike	51	24

- It was previously agreed that at the thrust level of 30,000 lbf for the MEM ascent engine, only the pump-fed mode would be considered. The dimensional requirements also favored the selection of pump-fed systems over pressure-fed systems. The use of the latter would not only entail larger dimensions than those shown above, but would also require substantial increases in engine and propellant tankage weight.
- Regenerative cooling was also selected as the most optimum type, from the standpoint of weight, durability, and performance, to be employed with pump feed. This cooling method was used for all propellant combinations except $C\ell F_5$/MHF-5, which was ablatively cooled.
- The Aerospike nozzle entails an approximate loss in performance of 1.5 percent of delivered specific impulse as compared to conventional bell

nozzles of the same expension ratio. However, the dimensional advantages of the Aerospike nozzle compensates for this minor loss.

The propulsion system parameters for the MEM ascent engine, as derived and refined from engine company data, are listed in Table 19. This table reflects the nominal parameters that were employed for each engine/propellant combination, including propellant characteristics, engine type, feed type, cooling systems, and engine size and weight.

Table 19

MEM PROPULSION-SYSTEM CHARACTERISTICS

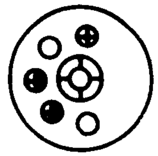
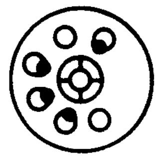
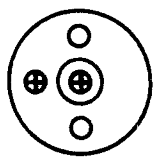
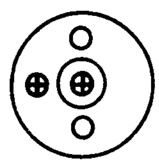
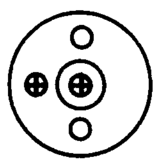
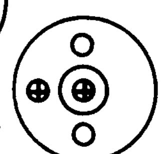
Propellant	Mixture Ratio	Chamber Pressure (psia)	ϵ	I_{sp} (sec)	Engine Wt. (lb)	Cooling
F_2H_2	13	750	75	463	440	Regenerative
O_2/H_2	6	750	100	449	520	Regenerative
$FLOX/CH_4$	5.7	750	75	400	440	Regenerative
OF_2/CH_4	5.3	750	75	406	460	Regenerative
$C\ell F_5$/MHF-5	2.4	750	100	336	475	Ablative

4.5 MEM ASCENT STAGE PERFORMANCE

The performance analysis was conducted in a manner similar to that for the Mars Orbiter. The propellant tanks for Stage II, within the main shell, are volumetrically limited by the vehicle configuration so that any propellant variation will be stored within Stage I. This is further complicated by the mission profile, which requires a ΔV of 13,800 ft/sec for the first burn and only 2,200 ft/sec for the second burn. Table 20 presents the propellant distribution within the two stages for the two burns for all propellants. The performance analysis indicated that, with the vehicle completely loaded, the O_2/H_2 system would deliver a ΔV of less than 15,000 ft/sec for

Table 20

MEM TANK AND PROPELLANT DISTRIBUTION

PROPELLANTS	CONFIGURATION	PROPELLANT CONSUMPTION	
		FIRST BURN	SECOND BURN
F_2/H_2		ALL OUTER TANKS PLUS 75% INNER TANKS	25% INNER TANKS
O_2/H_2		ALL OUTER TANKS PLUS 61% INNER TANKS	39% INNER TANKS
FLOX/CH_4		ALL OUTER TANKS PLUS 85% INNER TANKS	15% INNER TANKS
OF_2/CH_4		ALL OUTER TANKS PLUS 83% INNER TANKS	17% INNER TANKS
F_2/NH_3		ALL OUTER TANKS PLUS 84% INNER TANKS	16% INNER TANKS
ClF_5/MHF-5		ALL OUTER TANKS PLUS 86% INNER TANKS	14% INNER TANKS

○ OXIDIZER
⊕ FUEL

the given payload. Consequently, neither O_2/H_2 nor O_2H_2 with subcooled H_2 have system weights computed. The propulsion module total ascent stage weight and propellant load are given in Fig. 34. The F_2/H_2 system is the lightest weight system; the space storables are slightly heavier. The ClF_5/MHF-5 system is almost 50 percent heavier. Table 21 gives a detailed weight breakdown of the system weights.

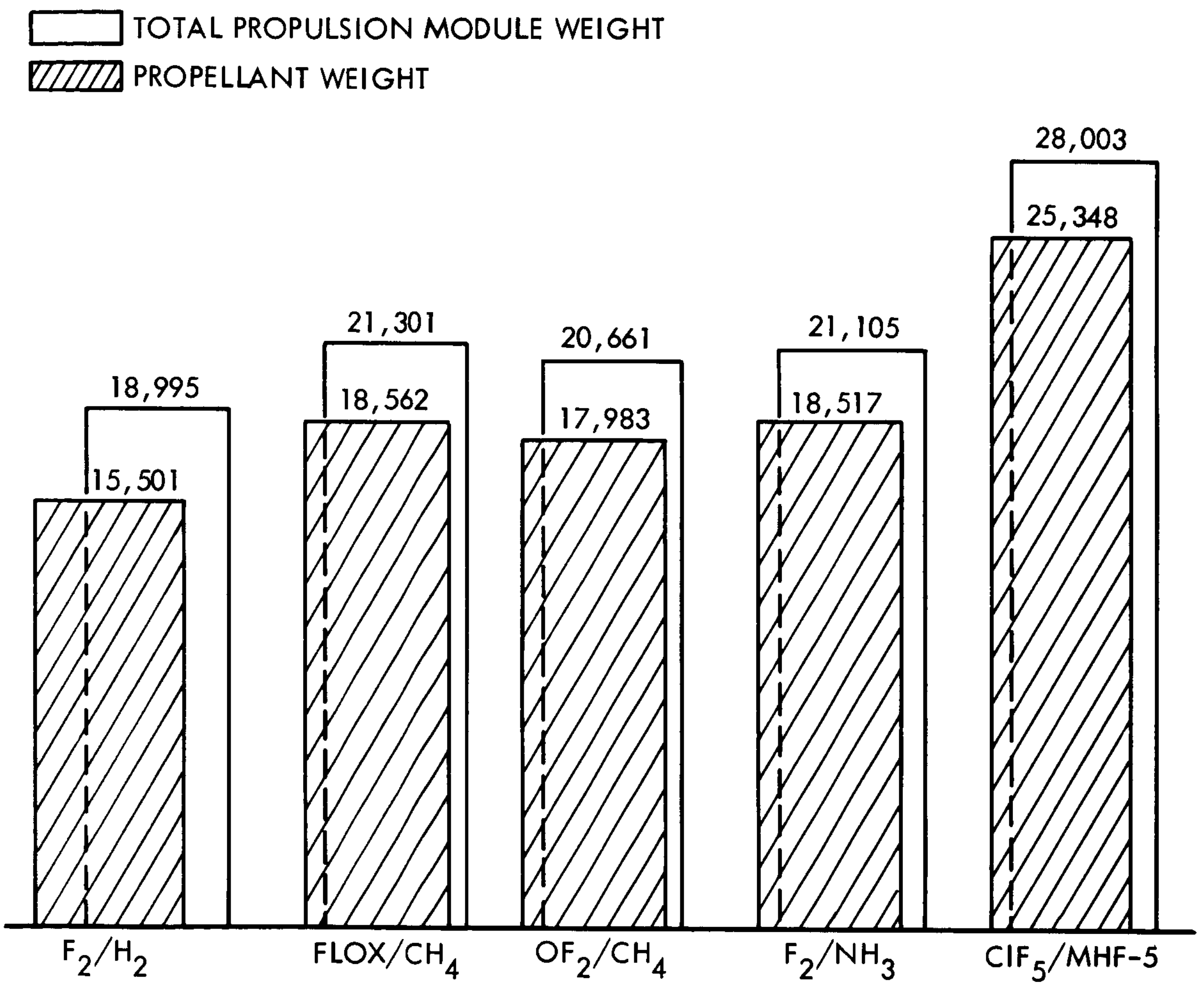


Fig. 34 MEM Weights for Propulsion Module

Table 21

MEM DETAILED WEIGHT BREAKDOWN

ITEM	WEIGHT (LB)									
	F_2/H_2		$FLOX/CH_4$		OF_2/CH_4		F_2/NH_3		ClF_5/MHF-5	
STAGE	I	II	I	II	I	II	I	II	I	II
STRUCTURE										
CENTRAL CORE	–	436	–	433	–	430	–	429	–	461
METEOROID PANELS	–	15	–	15	–	15	–	15	–	15
TANK SUPPORTS	184	57	116	57	96	56	105	56	121	56
THRUST STRUCTURE	–	80	–	80	–	80	–	80	–	80
PROPELLANT FEED ASSY.										
TANKS	468	181	275	184	272	186	258	187	262	190
VALVES	156	114	124	110	124	110	124	110	104	94
INSULATION	432	102	60	44	45	30	56	39	–	–
METEOROID PANELS	145	–	68	–	59	–	52	–	59	–
PRESSURIZATION	37	77	24	105	24	93	24	97	21	75
ENGINE SYSTEM	–	440	–	440	–	460	–	440	–	475
CONTINGENCY	33	150	16	147	14	146	14	145	14	144
RESIDUALS										
VAPOR WEIGHT	16	38	5	51	7	43	3	51	1	20
He-GAS	10	2	17	6	15	7	10	5	3	1
PROPELLANT	–	90	–	102	–	102	–	120	–	182
PERFORMANCE RESERVE	–	231	–	260	–	264	–	168	–	277
PROPELLANTS	12,016	3,485	12,218	6,344	11,378	6,605	11,302	7,215	15,048	10,300
STAGE WEIGHT	13,497	5,498	12,923	8,378	12,034	8,627	11,948	9,157	15,633	12,370
ASCENT STAGE WEIGHT	18,995		21,301		20,661		21,105		28,003	

4.6 SUMMARY OF MEM ASCENT STAGE ANALYSIS

The MEM provided an alternative to the Mars Orbiter because it had very restrictive design considerations and required propellant storage on the surface of Mars.

The design limitations were caused by the Apollo-shaped module and overall Aerobraker spacecraft and the mission profile. The MEM shape dictated the rigid propellant tank packaging requirements and also indicated that growth capability for the F_2/H_2 propellant was not possible and that O_2/H_2 could not be accommodated and still meet the mission requirements. The thermodynamic analysis indicated that the storage of propellants on the surface of Mars required an evacuated insulation system to obtain the required insulation effectiveness of multilayer insulation. The space limitation also affected the propulsion in that an Aerospike or other torroidal engine is required if a single engine is specified. The performance characteristics of this stage indicated that the lowest weight system was obtained with F_2/H_2 propellants.

Section 5
GROUND OPERATIONS

5.1 GROUND SUPPORT ANALYSIS

To assist in launch-pad propellant temperature control and to minimize boiloff and subcooling requirements, compartments requiring thermal conditioning should be supplied an atmosphere at the minimum temperature permissible. Because of the spacecraft equipment section temperature requirements, the minimum temperature would be on the order of 40° F (500° R).

The H_2 propellant tank insulation system requires a helium atmosphere to prevent condensation of gases on the tank and within the insulation. The space storables will require either dry nitrogen or dry air to prevent condensation of water vapor on the tank or within the insulation. The earth storables will not require special measures other than those dictated by the equipment section.

To determine the operational support required, an analysis was conducted to determine the effect of the prelaunch environment on the various propellants. The insulation thicknesses used for the tanks are those selected during the optimization of the Mars Orbiter pump-fed systems. The gas introduced into the shroud has been assumed to be at 500° R. To determine the heat gains into the propellant, effective thermal resistances were determined between the propellant at its normal boiling temperature and the environment of 500° R. The insulation conductivity was assumed to be equal to the purge gas conductivity at an average between the propellant temperature and 500° R. A convection coefficient of 1 Btu/hr-ft^2- R was assumed over the entire outer surface of the insulation. Boiloff rates, vapor vent rates, and the amount of subcooling required (to compensate for ascent heating) as a function of the time between liftoff and end of propellant topping were determined for each propellant.

The results of the prelaunch requirements are presented on Table 22 for the cryogenic and space-storable propellants. All values are for the entire spacecraft (all tanks). The preliminary ground-support equipment requirements for the various propellant combinations also are presented.

A brief analysis (see Volume III) was conducted to determine the propellant temperature rise during the ascent phase. It takes approximately 200 sec to vent the insulation; therefore, a high insulation conductivity was used (about an order of magnitude lower than the gas value) for this period with an increased temperature difference to determine the propellant temperature rise. Assuming that all the energy is absorbed by the propellant mass, the temperature rise is on the order of 0.05° F or less, which is negligible and can be accounted for by subcooling a very slight amount. Even if the ascent temperature rise were considered to occur for a 1-hr period with the same conductivity and high insulation temperature, the propellant temperature increase is 0.5° F or less, which can be compensated for by subcooling.

5.2 GROUND SUPPORT EQUIPMENT AND OPERATIONS

To properly evaluate candidate propulsion systems for specific missions, all significant parameters must be considered. This includes ground operational constraints, which can be very important considerations, especially when applicable ground equipment and operational procedures are already in existence from previous vehicle launchings and funding. Overall program costs can be greatly affected by whether or not a system is compatible with this existing equipment.

Pad 39 at Kennedy Space Center is assumed to be similar to that which will be available for a Mars mission launch. There will be a strong desire to limit modifications to the launch pad to a minimum in order to fully exploit the advantages of propellant combinations that are already being used. The present system of loading probably will be adhered to if possible. Saturn V loading flow data are given on page 118 for reference.

Table 22

PRELAUNCH TANKAGE THERMAL EFFECTS
(Mars Orbiter, Pump-Fed, 205-Day Mission)

PROPELLANT	PRELAUNCH THERMAL CONDITIONING	HEAT INPUT PER VEHICLE (BTU/HR)	VENTING		NONVENTING		
			BOILOFF (LB/HR)	VENTED VAPOR (FT^3/HR)	NONVENTED TEMP RISE (^oR/HR)	NONVENTED OPERATING PRESS. RISE (PSI/HR)	NONVENTED HOLD WEIGHT PENALTY (LB/HR)
F_2 H_2	A	3,200 5,150	43.4 26.7	416 2,900	1.6 4.6	1.5 10.0	0.75 17.5
O_2 H_2	A	6,800 8,160	73.6 42.3	410 4,600	3.1 3.7	3.2 8.0	0.96 27.6
FLOX CH_4	B	2,500 3,000	31.0 13.6	328 310	1.2 3.4	1.0 2.3	0.48 0.39
OF_2 CH_4	B	2,500 3,100	30.8 14.0	208 318	1.6 3.3	1.1 2.1	0.46 0.37
F_2 NH_3	B	1,726 156	23.4 0.2	224 5.2	0.9 0.1	0.9 0.03	0.43 NEGLIGIBLE
N_2O_4 A-50	C	NEGLIGIBLE	NEGLIGIBLE		NEGLIGIBLE		
ClF_5 MHF-5	C	NEGLIGIBLE	NEGLIGIBLE		NEGLIGIBLE		

A. DRY N_2 INTO SHROUD AT 40°F AND 10 TO 50 LB/MIN, PLUS HELIUM PURGE BAG FOR H_2 TANK.
B. DRY N_2 OR AIR INTO SHROUD AT 40°F AND 10 TO 50 LB/MIN.
C. REQUIREMENTS COMPATIBLE WITH THE EQUIPMENT SECTION (40° TO 70°F)

H_2		
	S-II 120 Level (gpm)	S-VIB 200 Level (gpm)
Precool	1,000	500
Fast Fill	10,000	3,000
Slow Fill	1,000	500
Replenish	0 to 500	0 to 200
Drain	6,670	4,500

O_2			
	S-IC (gpm)	S-II (gpm)	S-IVB (gpm)
Precool	1,500	500	500
Fast Fill	10,000	5,000	1,000
Slow Fill	1,500		300
Replenish	0 to 500	0 to 200	60
Drain	7,900	3,300	1,370

N_2O_4
70 gpm fill — level 220 (from transfer unit) 6 gpm fill — (RCS only) 70 gpm return — (to ready storage unit)

Hydrazine
70 gpm (from ready transfer unit)

The facility currently consists of remote storage dewars, ready storage units, transfer and conditioning units, toxic vapor disposal units, thermal conditioning systems, purge systems, and high-pressure pneumatic supplies. These systems are used during normal ground operations and during emergency situations. The operations to be described in this section are method of transfer, spill disposal, vent gas disposal, and tank venting.

5.2.1 Transfer

Transfer from delivery vehicle to main storage can be accomplished by either pressure or pump methods. Pressure drop can be very low if necessary since little or no head due to elevation will be encountered. Main storage should be remote from the launcher/ umbilical tower for all propellants. This reduces hazards such as spills, fire, and inadvertent toxic or hazardous vapor venting in the vicinity of the tower. It also allows spearation of fuel and oxidizer storage.

Transfer to the vehicle will be by pump, except for hydrogen. Pump transfer is desirable for most of the propellants because a fairly large head is developed owing to the height to which the fluid is transferred. If pressure transfer were to be used, the main storage container would have to be pressurized to greater than 200 psi in some cases. With a pump transfer system, the only item that would be subjected to such high pressure would be the transfer line from the base of the tower to the vehicle. Hydrogen, however, has a very low density, does not build up a large head (only 5 to 6 psi), and can be transferred conveniently by pressure.

If any of the spacecraft stage propellants are also to be loaded in one of the boost stages, the main supply can be used to load the upper stage. If insufficient pressure is available, a boost pump can be installed in the system. Recommended fill rates are as follows:

Slow Fill	10 gpm
Fast Fill	50 gpm
Replenish	0 to 10 gpm (space storable and cryogen only)
Drain	30 gpm

These rates apply to all tankage except hydrogen. Hydrogen rates could be approximately twice as great because there is only one large-volume tank instead of multiple tankage.

For those propellants peculiar only to the spacecraft, a system similar to the existing N_2O_4 or hydrazine systems should be used. This setup consists of a remote storage dewar, ready storage unit, transfer and conditioning unit, toxic disposal unit, and associated plumbing. The ready storage unit is located on the launcher/umbilical tower base, and is used to hold propellant ready for transfer. It also is the receiver if necessary to drain the tanks. The transfer and conditioning unit is necessary to bring subcooled (earth storable) propellants within temperature limits prior to transfer to the vehicle. A toxic disposal unit is necessary to either change the chemical composition of vented vapors or to dilute the vapors to an acceptable level. More specific information related to the individual propellants is presented in the following paragraphs.

Fluorine, FLOX, and OF_2. An Apollo LM transfer system might be modified for fluorine service. Either the N_2O_4 or A-50 system would have approximately the proper flowrate. Adjustment in pump output pressure or transfer-line flow resistance could be accomplished without much difficulty. Material compatibility would be of major concern in such a conversion. Most static and dynamic seals would have to be replaced, valves and filters would have to replaced, and some of the piping would be incompatible. Also of major concern would be the low-temperature shrinkage involved and the vacuum-jacket installation necessary to reduce heat transfer during flow. It would, therefore, be advisable to design and install an entirely new system for fluorine. Boiloff should be minimized during transfer to reduce the problem of vapor disposal during fill. A ready storage unit that is thermally insulated and cooled could be used to advantage to reduce heat transfer. To further reduce heat leak into the system, the transfer line can be jacketed with O_2 or N_2. Emergency drain will be into storage unit.

Hydrogen Transfer. Transfer of hydrogen could easily be accomplished by branching off from the main supply. Since this is pressure transfer, no problem of flow capacity will be encountered. The branch can be bypass-orificed to get the desired fill and replenish flowrates. The boost stage should be loaded prior to the spacecraft so that the large volumes of hydrogen boiloff vapor can be vented through the larger tank vent. If hydrogen as a fuel for the boost stage has been abandoned, which is unlikely, a completely new pressure transfer system will have to be installed for the spacecraft.

Oxygen Transfer. The boost stage oxygen loading system could be modified for transfer downstream from the pumps to the spacecraft. Sufficient pump pressure should be available to overcome the additional head required for the higher elevation. Pump output for the oxygen replenish system is 260 psi. If this is insufficient, a boost pump can be installed in the system. The output of the main system pump is considerably greater than that required for the spacecraft slow-fill; therefore, a bypass orifice can be installed for flow control. If the main pump overloads and does not have an overpressure relief capability for sustained operation during the spacecraft fill, a low-capacity pump must be installed in the branch. This pump should be located as close to the tower as possible, but still be near ground level. This would allow liquid to be drawn directly from the dewar by the small pump. The NPSH required would be available even with very low dewar pressure because little, if any, potential energy is required to get to the pump elevation, and only 0.5 psi line pressure drop from the dewar to pump will occur during fill.

Methane Transfer. Conversion of an Apollo LM transfer system could be relatively easy for liquid methane service. Little or no trouble would be encountered from the material compatibility standpoint. Some seals may have to be replaced and bellows installed because of low-temperature shrinkage. Vacuum jacketing will have to be installed for all transfer lines and storage containers. The ready storage unit need not be used.

Ammonia Transfer. Same comments as CH_4 except that there probably will be no low-temperature shrinkage problems. No refrigeration is required for main storage.

N_2O_4, A-50 Transfer. There will be no problem in using the Apollo LM loading system directly.

ClF_5 Transfer. Little information is available on ClF_5, but it is similar to ClF_3. Generally, it is less reactive than ClF_3, and, therefore, procedures recommended for ClF_3 will be used. The Apollo LM transfer system could be converted easily for ClF_5 service. Little trouble would be encountered from the material compatibility standpoint. The entire system must be passivated. ClF_5 is an earth storable and will present no thermal compatibility problems. Storability of ClF_5 is good. It is thermally stable and shock insensitive. Long-term exposure to moisture will change the composition, increasing the ClF_3 content and reducing the ClF_5 content. The liquid will become corrosive when moisture is present.

MHF-5 Transfer. An Apollo LM A-50 transfer and storage system could be used directly for MHF-5 service.

5.2.2 Spill Disposal

Inadvertent spills are always a hazard, and must be handled in a manner that will keep danger to equipment and personnel to a minimum. Although preventative measures are taken to preclude such an occurrence, the possibility still exists. Accidental damage to equipment, contaminants in liquids, human error in operational procedures, etc., cannot be completely eliminated; therefore, the system must reduce spill hazard to a minimum.

Spills expose personnel and equipment to the dangers of explosions, fire, and toxic and/or corrosive liquids and vapors. It is imperative that personnel be trained in the handling and safety procedures for the materials in use. This alone, however, is insufficient to minimize the dangers involved. The equipment must be designed with built-in safety measures, including drain troughs, spill basins, water dilution, heat sinks, and chemical neutralizers.

Fluorine, FLOX, and OF_2 Spill Disposal. A drain trough to transport these propellants to a somewhat remote spill basin should be provided. The basin need only be removed from directly underneath the tower. This will allow corrosive vapors to rise without direct impingement on the vehicle and launch equipment. All basins and troughs should be constructed of concrete. The spill basin can be lined with limestone for reaction. A water deluge system could be employed. A system of directional control gates may be necessary if the fluorine and fuel basins are incompatible. If a fire develops, the reaction is likely to be so rapid that no attempt can be made to extinguish the flame. After the fluorine-fed fire has subsided and the fluorine has been consumed, or has evaporated, efforts should be directed toward reducing secondary fires. Spills may be handled by remote application of water fog, fine water spray, or soda ash to promote smooth, rapid combustion of the fluorine. These problems and solutions also apply to FLOX and OF_2.

Hydrogen Spill Disposal. The existing spill disposal system will be more than adequate. Crushed rock should be used in the basin to increase the exposed surface area of the basin and its heat-sink capability. Hydrogen can be disposed of by vaporization, which will be accelerated by the increased heat sink. Hydrogen gas is extremely flammable, and a serious fire hazard always exists when hydrogen-gas vapors are in the area. With no impurities present, hydrogen burns in the air with an invisible flame. Extreme measures should be taken to prevent spark discharge. A hydrogen fire can be effectively controlled with heavy concentrations of water, CO_2, or stream.

Oxygen Spill Disposal. The existing spill system will be adequate. Disposal will take place by natural vaporization. Crushed rock will help accelerate vaporization. If a fire develops, all flow should be shut off. For large spill fires, wait until the oxygen has evaporated, and then use Class B fire extinguishing methods on remaining fires. Small spill fires may be extinguished directly using large quantities of water. The potential for an explosion is always present with spilled oxygen.

Methane Spill Disposal. Spill basin design for hydrogen is adequate. The spill should be deluged with water or water spray to reduce fire hazard. Fire hazard is not as great as with hydrogen. If fire does develop, the flame will be visible and can be extinguished with water, CO_2, or steam.

Ammonia Spill Disposal. The existing spill basin design is adequate. Since this fuel will be used with fluorine, a directional control gate may be necessary to separate spill basins (refer to discussion on fluroine) if two basins are required. Water deluge is required to reduce fire, explosion, and toxic hazards.

The flammability range of ammonia is at higher concentration than for hydrocarbons, but large spills will present a fire hazard. Ammonia fires are very difficult to extinguish. Water fog is recommended for ammonia fires because it cools the burning surfaces and reduces the vapor pressure by absorption and dilution. Large quantities are required. The explosion hazard of ammonia is relatively low compared to hydrogen.

N_2O_4 Spill Disposal. The existing spill basin is adequate. The area should be deluged with water to reduce the fire hazard; however, water will accelerate fuming. Nitrogen tetroxide supports combustion; if fire is present, deluge with water. Continued application of large quantities of water will eventually dilute the oxidizer so that combustion is no longer supported. Remaining air-supported fires may be extinguished by ordinary means.

Aerozine-50 Spill Disposal. Use present spill basin. Area should be deluged with water to reduce the fire hazard. If fire is present, water is the safest and most effective agent to use. Only water is recommended for oxidizer-supported fires if it is compatible with the oxidizer. If the fire is air-supported and it is a small spill, bicarbonate-base (power type) agents are the most effective. Water fog or carbon dioxide may also be used. If the spill is large and air-supported, only large amounts of coarse spray water are recommended. The water fog, CO_2, and bicarbonate methods are subject to backflashes and explosive reignitions. The A-50 propellant readily forms an explosive mixture with air which can be ignited by a spark or flame.

ClF_5 Spill Disposal. A spill basin design similar to the fluorine system may be necessary. Currently, ClF_5 will be handled in the same manner as ClF_3. Powered carbonate or bicarbonate, water spray, ammonia, or carbonate solutions should be used to decontaminate spillage. It may also be disposed of in an isolated area by piping it to an evaporator basin containing crushed rock. Fires may be controlled using water fog or spray, which will smooth the reaction. Complete coverage of the area will minimize the evolution of hydrogen fluoride and chlorine fumes – ClF_5 may react vigorously with water and most combustible substances at room temperature. In addition, it reacts strongly with silicon-containing compounds and can support continued combustion.

MHF-5 Spill Disposal. The spill basin design should be the same as for A-50. Spills should be deluged with water to reduce fire hazard. The same general comments as A-50 apply for fire fighting. MHF-5 is composed of 55 percent MMH, 20 percent N_2H_4, and 19 percent $N_2H_5NO_3$. Upon vaporization the hydrazine nitrate content increases and the mixture becomes shock sensitive.

5.2.3 Vent Gas Disposal

Vent gases must be disposed of for two primary reasons: to reduce the potential for fire or explosion and to eliminate the toxic and corrosive dangers. Vent gases are usually routed through a pipe to an area remote from the launch vehicle and personnel. It is then free-vented to the air or burned.

Fluorine, FLOX, and OF_2 Gas Disposal. These gases are extremely toxic and can cause severe burns and pulmonary edema. The total mass of vented vapors should be kept to an absolute minimum. All gas should be piped to a remote vapor disposal unit. This unit may contain charcoal to reduce the fluorine, FLOX, or OF_2 content sufficiently if small quantities are vented. Fluorine gas may also be combined with propane during a burning process. Hydrogen fluoride gas will be a by-product of combustion. This gas is also toxic and may be scrubbed through charcoal. The latter process may be more convenient since less charcoal is required. The container for charcoal need only be an open concrete pit that free-vents the gases to the atmosphere. Periodic replacement of the charcoal is necessary since it will be consumed during combustion.

Hydrogen Vent Gas Disposal. Hydrogen vapor in the quantities used during loading can be free-vented to the atmosphere through a remote standpipe. It can be burned if necessary in the burn pond provided for the S-IVB.

Oxygen Vent Gas Disposal. Oxygen can be free-vented to the atmosphere.

CH_4, NH_3, N_2O_4, MHF-5 Vent Gas Disposal. Vapors can be free-vented to the atmosphere in quantities formed during loading of spacecraft or run through a vapor disposal unit such as provided for the Apollo LM.

ClF_5 Vent Gas Disposal. This gas should be treated in the same manner as fluorine. It is highly toxic and should be neutralized. Very little vapor will be formed because ClF_5 is earth storable.

5.2.4 Tank Venting

Procedures for tank venting and cap-off can significantly affect the performance of a space vehicle. This is mostly true of space-storable and cryogenic propellants because of the low temperatures of the propellants relative to the ground environment. Heat will be absorbed because the propellant temperatures are lower than ground temperatures. This heat can either go directly into boiloff or to raise the temperature of the propellant. This is dependent upon whether or not the tanks are vented.

Nonvented Tanks. As soon as the tanks are capped-off, the tank pressure can rise. Any boiloff will make the tank pressure rise. If boiloff were to occur without heat absorption by the bulk, the liquid would become subcooled and the capacity to absorb heat would be increased. For analysis purposes it was assumed that there is enough convection in the propellant to maintain vapor pressure and liquid saturation pressure in equilibrium. The result is that a negligible amount of heat goes to boiloff, with nearly all heat being absorbed by the propellant. Therefore, the longer the period of nonvent, the higher the saturation pressure will be at liftoff. The final tank pressure at the end of the mission is directly affected by the liftoff saturation pressure. The difference between final and liftoff saturation pressure is nearly constant for any reasonable initial propellant condition. Therefore, the higher the saturation pressure at liftoff, the greater the final tank pressure at the end of the mission. This increased pressure results in an increase in system weight. The weight of pressurant required is increased and tank structural weight may increase. In most cases, the minimum manufacturing gage is greater than that required for the increased pressures. The notable exception to this is the hydrogen tanks. To evaluate the effect of nonvent hold time, the system weight increases for the Mars orbiter were calculated. The results are given in the following table. The weight increase in lb/hr is the system weight penalty required for each hour of hold capability designed into the vehicle.

System	F_2	H_2	O_2	H_2	FLOX	CH_4	OF_2	CH_4	F_2	NH_3	N_2O_4	A-50	ClF_5	MHF-5
Weight Penalty (lb/hr)	0.75	17.5	0.96	27.6	0.48	0.39	0.46	0.037	0.43	0	0	0	0	0

A ground refrigeration system for the cryogenic propellants could extend the ground nonvent hold time indefinitely without a time-dependent weight penalty. Fixed weight penalty associated with the refrigeration system would be a function of individual system design.

Earth-storable propellants do not incur a weight penalty, as can be seen. Only the hydrogen tanks should definitely not be capped off until just prior to liftoff. Recommended maximum hold times in the nonvented mode are shown in Table 23.

Vented Tanks. The cryogens and space-storable propellants can be left vented with umbilicals intact until just prior to liftoff. This is the current practice with the Saturn V. Venting is recommended over the nonvented mode except for the earth-storable propellants because there is no system weight penalty with hold time. The umbilicals would be removed upon initial vehicle motion. Venting capability and draining capability would be possible at any time prior to liftoff.

Earth-storable propellants can be loaded several days in advance of liftoff. There would be no need to drain and the umbilicals could be disconnected. It if it necessary

Table 23

PROPELLANT GROUND HANDLING

Propellant	Transfer Mode	Transfer Equipment	Spill Disposal	Hazards	Vent Gas Disposal	Recommended Tank Cap-Off
F_2	Pump	Recommend entirely new transfer system Ready storage unit should be used for abort drains. Ready transfer and conditioning unit may be necessary for subcooling.	Drain to remote basin if possible. Concrete with limestone for reaction. Water deluge should be used for flushing. Separate basin for final spill may be necessary.	Very toxic. Fire reaction may be very rapid. F_2 should evaporate before trying to put out fire. Use water, fog, spray, or soda ash.	Toxic vapor unit necessary. May contain charcoal for scrubbing or burn with propane in conjunction with charcoal. Boiloff should be minimized during fill.	0 to 10 hr before lift-off
H_2	Pressure	Branch off main supply.	Use existing spill basin. Should have crushed rock bed.	Explosion and fire hazard always present. Control fires with water, CO_2 or stream. Flame outer edge not visible.	Use existing burn pond or free-vent through standpipe.	Just prior to lift-off
O_2	Pump	Branch off of existing main supply and small pump to system.	Use present spill facilities. Dispose by evaporation. Rock bed accelerates evaporation.	Allow O_2 to evaporate before extinguishing. Explosion hazard always present with spill. Non-toxic.	Free-vented to atmosphere.	0 to 10 hr before lift-off
H_2	Same as H_2 above					
FLOX	Same as F_2 above					
CH_4	Pressure	Convert Apollo L. M. transfer system. Will need to install new seals and bellows. Jacketed lines necessary.	Use existing spill basin. Deluge with water or water spray to reduce fire hazard.	Fire hazard not as great as for hydrogen. Water, CO_2, or steam for fire control. Low toxicity.	Vapors can be free-vented or burned in air.	0 to 10 hr before lift-off
OF_2	Same as F_2 above					
CH_4	Same as CH_4 above					
F_2	Same as F_2 above					
NH_3	Pump	Convert Apollo L. M. transfer system. May need new seals. Vacuum jacket not necessary. May use transfer and conditioning unit.	Present spill basin adequate. Directional control gates to separate from a remote F_2 basin may be necessary. Water deluge.	Low explosion hazard compared to hydrogen. Flamability concentration greater than hydrocarbons. Fire very difficult to extinguish. Use water, fog. Relatively low toxicity.	Vapors can be free-vented.	Anytime after top-off
N_2O_4	Pump	Use existing system for Apollo L. M.	Use present spill basin. Deluge with water.	Will support combustion. Water deluge to fight fire. Toxic.	Vapors can be free-vented or burned in a disposal unit.	Anytime after top-off
A-50	Same as N_2O_4 above					
CLF_5	Pump	Convert Apollo L. M. transfer system.	Basin design similar to F_2 is recommended. Use powdered carbonate, water spray, or ammonia to decontaminate.	Treat as fluorine. Highly toxic. Use water fog or spray on fires. Reacts with silicone compounds.	Treat as fluorine. Use a disposal unit with reactant.	Anytime after top-off
MHF-5	Same as N_2O_4 above					

to drain, both the fill and vent disconnects will have to be remated. Proper thermal conditioning will be accomplished by the pad environmental control system. Purge gases in the vehicle cavities will be preconditioned to maintain temperature within tolerance. No venting will be necessary while the tanks are sealed off.

Section 6
SENSITIVITY ANALYSES – MARS ORBITER

An assessment was made to determine the system effects of varying several of the design parameters for the Mars Orbiter pump-fed vehicle. The parameters investigated, including mission length, surface coatings, meteoroid flux, etc., are discussed in the following paragraphs.

6.1 MISSION LENGTH

The first investigation was made by extending the mission duration to a total of 300 days from 205 days. All mission sequences and velocity steps were kept constant, except for the interplanetary transit, which was extended by 95 days. The tank design pressure and insulation thickness for the baseline sun on tank, nonvented systems are presented in Table 24. The indicated differences were relatively small; therefore, a further analysis was initiated. The three-burn mission was substituted for the standard four-burn mission. This simplified the thermodynamic optimization significantly. Three mission lengths were investigated: 195 days, 290 days, and 650 days. The 195- and 290-day missions are the previously discussed Mars Orbiter mission, and the 650-day mission utilizes the same spacecraft on a trip to Jupiter. The tank operating pressure and insulation thickness for the various propellant combinations are shown in Table 25. In addition, the propulsion module weights for all of the cases are shown. The longer transit Mars mission represents a small weight penalty when either cryogens or space storables are used. For the 650-day Jupiter mission, there is only a slight penalty for the cryogens and essentially no penalty for the space storables. For the earth storables, there is an interim tank pressure rise that is well below the tank minimum gauge limit. The pressure also drops toward the end of the mission so that the pressurant residuals do not affect the system weights.

Table 24

MARS ORBITER SENSITIVITY TO MISSION DURATION
(205-Day vs. 300-Day Mission, Sun-On-Tank, Nonvented, Pump Fed)

Propellant	Surface Finish (α/ϵ ratio)	205-Day Mission: Max. Oper. Pressure (psia)	205-Day Mission: Insulation Thickness (in.)	205-Day Mission: Propulsion Module Weight (lb)	300-Day Mission: Max. Oper. Pressure (psia)	300-Day Mission: Insulation Thickness (in.)	300-Day Mission: Propulsion Module Weight (lb)	Weight Change (%)
F_2	0.05/0.80	40	1-1/8	7,327	50	1-1/4	7,442	1.6
H_2	0.05/0.80	130	4-5/8		150	5-3/4		
O_2	0.05/0.80	58	3/4	8,477	80	1	8,721	2.9
H_2	0.05/0.80	96	4-5/8		114	5-1/4		
FLOX	0.05/0.80	57	1-1/2	7,968	75	1-5/8	8,023	0.7
CH_4	0.05/0.80	107	3/4		140	1		
OF_2	0.05/0.80	45	1-1/8	7,874	56	1-1/4	7,921	0.6
CH_4	0.05/0.80	107	3/4		140	1		
F_2	0.05/0.80	59	1-1/8	7,984	74	1-5/8	8,009	0.3
NH_3	0.60/0.91	< 15	1/4		< 20	1/4		

Note: N_2O_4/A-50 and ClF_5/MHF-5 are relatively insensitive to mission duration.

Table 25

MARS ORBITER VEHICLE – SENSITIVITY TO MISSION DURATION
(Sun-On-Tank, Pump-Fed, Nonvented, Optimum α/ϵ)

PROPELLANT	195-DAY MISSION			290-DAY MISSION			650-DAY JUPITER MISSION		
	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE (LB)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)
F_2	80	*	7,080	100	5/8	7,224	110	1/2	7,299
H_2	120	4-1/4		140	5-1/4		190	5-3/4	
O_2	100	*	8,206	160	1/2	8,382	168	1/2	8,615
H_2	86	3-3/4		106	4-1/4		174	4-1/2	
FLOX	150	5/8	7,796	175	5/8	7,811	175	3/4	7,809
CH_4	164	*		180	5/8		210	1/2	
OF_2	120	*	7,723	123	1/2	7,750	133	*	7,730
CH_4	164	*		180	5/8		210	1/2	
F_2	74	1/2	7,825	183	5/8	7,858	183	3/4	7,865
NH_3	<15	**		<20	**		<30	**	
N_2O_4	<15	**	9,527	<20	**	9,527	<30	**	9,527
A-50	<15	**		<20	**		<30	**	
ClF_5	<15	**	9,218	<20	**	9,218	<30	**	9,218
MHF-5	<15	**		<20	**		<30	**	

*MINIMUM VALUE 1/2-IN. SUPERINSULATION.
**MINIMUM VALUE 1/4-IN. FOAM.

6.2 SURFACE COATINGS

To assess the sensitivity to surface characteristics, the various propellant combinations were evaluated with both silver and aluminum-backed optical surface reflectors (OSR), white thermatrol paint, and white skyspar paint. Table 26 presents the system weights resulting from using these coatings for the sun on tank vehicle configuration. For the cryogens and space storables, the silver-backed OSR yielded the lightest weight system, and for the earth storables, white paint provided the lightest weight system.

6.3 METEOROID FLUX

The meteoroid flux was increased by a factor of ten to evaluate the effect of a very severe change in the environment. This effect more than doubled the actual weight of the meteoroid shield. The effects on system weight are shown in Table 27.

It is apparent that a change in flux of this magnitude has a severe effect on system weight, but it affects all systems uniformly and there is no displacement of one propellant combination by another propellant combination.

6.4 SPECIFIC IMPULSE

The effect of varying the specific impulse was also evaluated. The specific impulse values used are nominal, but somewhat optimistic according to some sources, so that a sensitivity analysis seems appropriate. An assessment was made of a ±3 percent change in specific impulse. The actual values of specific impulse used and the propulsion module weights are compared in Table 28 with the basic system. Even when comparing the poorest performing space storable, F_2/NH_3 , at the low specific impulse with the best performing earth storable, ClF_5/MHF-5, at the high specific impulse there is a greater than 10 percent spread in specific impulse, which is also reflected in propulsion module weight.

Table 26

MARS ORBITER SENSITIVITY TO TANK SURFACE FINISH
(Sun-On-Tank, Pump-Fed, Nonvented, 205-Day Mission)

Propellant	OSR Silver Backed α/ϵ = 0.05/0.80			OSR Aluminum Backed α/ϵ = 0.10/0.80			White Thermatrol Paint (Degraded) α/ϵ = 0.30/0.95			White Skyspar Paint (Degraded) α/ϵ = 0.60/0.91		
	P	T	WT	P	T	WT	P	T	WT	P	T	WT
F_2	40	1-1/8	7,238				75	2	7,327			
H_2	130	4-5/8					150	4-3/4				
O_2	58	3/4	8,477				85	1	8,576			
H_2	96	4-5/8					115	4-3/4				
FLOX	57	1-1/2	7,968	65	1-5/8	7,997	70	2	8,063			
CH_4	107	3/4		110	7/8		160	1-1/8				
OF_2	45	1-1/8	7,874	52	1-1/4	7,887	58	1-1/2	7,943			
CH_4	107	3/4		110	7/8		160	1-1/8				
F_2	59	1-3/8	7,993	68	1-1/2	8,006	70	2	8,030			
NH_3	<15	MIN		<15	MIN		16	MIN				
N_2O_4	<15	1	9,639	<15	3/4	9,615	<15	1/2	9,567	<15	MIN	9,535
A-50	<15	1-3/8		<15	1-1/4		<15	3/4		<15	MIN	
ClF_5	<15	MIN	9,227	<15	MIN	9,220	<15	MIN	9,220	<15	MIN	9,220
MHF-5	<15	3/8		<15	1/4		<15	MIN		<15	MIN	

P = Tank maximum operating pressure (psia)
T = Insulation thickness (in.)
WT = Propulsion Module Weight (lb)

Table 27

MARS ORBITER METEOROID FLUX SENSITIVITY
(Sun-On-Tank, Pump-Fed, 205-Day Mission, Optimum α/ϵ)

Propellant	Propulsion Module Weight (lb)		Weight Change (%)
	Basic Flux	10 × Basic Flux	
F_2/H_2	7,238	7,503	3.7
O_2/H_2	8,477	8,885	4.8
FLOX/CH	7,968	8,141	2.2
OF_2/CH_4	7,874	8,047	2.2
F_2/NH_3	7,993	8,193	2.5
N_2O_4/A-50	9,535	9,811	2.9
ClF_5/MHF-5	9,220	9,476	2.8

Table 28

MARS ORBITER SENSITIVITY TO SPECIFIC IMPULSE
(Sun-On-Tank, Pump-Fed, Nonvented, 205-Day Mission, Optimum α/ϵ)

Propellant	-3%			Nominal		+3%		
	I_{sp} (sec)	Propulsion Module Weight (lb)	% Wt Change From Nominal	I_{sp} (sec)	Propulsion Module Weight (lb)	I_{sp} (sec)	Propulsion Module Weight (lb)	% Wt Change From Nominal
F_2/H_2	453.96	7,455	3.0	468	7,238	482.04	7,035	-2.8
O_2/H_2	437.47	8,757	3.3	451	8,477	464.53	8,214	-3.2
$FLOX/CH_4$	397.70	8,231	3.3	410	7,968	422.30	7,721	-3.2
OF_2/CH_4	397.70	8,134	3.3	410	7,874	422.30	7,638	-3.1
F_2/NH_3	395.76	8,257	3.3	408	7,993	420.24	7,753	-3.1
N_2O_4/A-50	324.95	9,865	3.5	335	9,535	345.05	9,226	-3.3
ClF_5/MHF-5	331.74	9,544	3.5	342	9,220	352.26	8,925	-3.3

6.5 INSULATION CONDUCTIVITY

Baseline insulation conductivities of 2.5×10^{-5} Btu/hr-ft-°R were used for hydrogen tanks; 5×10^{-5} for oxygen, fluorine, and the space storables; and 10×10^{-5} for NH_3 and the earth storables. For the degraded insulation case, the conductivity was multiplied by a factor of two for all propellant combinations. Table 29 lists the operating pressure, insulation thickness, and propulsion module weight for the baseline case and the systems with assumed degraded insulation. The largest effect is 5 percent on the O_2/H_2 system, whereas the other systems indicate changes varying from 0.6 percent for OF_2/CH_4 to 3.5 percent for F_2/H_2.

6.6 PROPELLANT INITIAL CONDITION

An analysis also was made of the effect of the initial condition of hydrogen and the venting of hydrogen for the cryogenic systems. The investigation was made with the sun on tank orientation comparing saturated, triple-point, 50-percent slush hydrogen and venting the hydrogen. In the vented hydrogen case the oxidizer is cooled by passing the vented hydrogen through the oxidizer tank.

Table 30 lists the operating pressure, insulation thickness, and propulsion module weights for the cases studied. By using either subcooled hydrogen or slush hydrogen, the tank weight and insulation thickness are reduced, which lower the system weight. By venting the boiloff, the hydrogen and oxidizer pressures and the hydrogen and oxidizer insulation thicknesses were reduced, yielding the lowest propulsion module weights of the cases studied. The initial weight for the vented cases include the boiloff weights listed in the table.

6.7 VEHICLE ORIENTATION

The effect of orienting the vehicle so that its propellant tanks are exposed to the sun or shielded from the sun can be significant in terms of insulation thickness, operating pressure, and system weight. Table 31 presents these data. It is significant that the hydrogen tank pressure can be reduced from 130 to 80 psi, the insulation thickness

Table 29

MARS ORBITER SENSITIVITY TO INSULATION CONDUCTIVITY
(Sun-On-Tank, Pump-Fed, Nonvented, 205-Day Mission, Optimum α/ϵ)

PROPELLANT	BASELINE				DEGRADED				WEIGHT INCREASE (%)
	INSUL CONDUCT., K (BTU/HR-FT-°R)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)	INSUL CONDUCT., K (BTU/HR-FT-°R)	OPER PRESS. (PSIA)	INSUL THICKNESS (IN.)	PROPUL MODULE WT (LB)	
F_2	5×10^{-5}	40	1-1/8	7,238	10×10^{-5}	58	1-3/4	7,492	3.5
H_2	2.5×10^{-5}	130	4-5/8		5×10^{-5}	187	6-1/4		
O_2	5×10^{-5}	58	3/4	8,477	10×10^{-5}	65	1-1/2	8,910	5.1
H_2	2.5×10^{-5}	96	4-5/8		5×10^{-5}	140	6-3/8		
FLOX	5×10^{-5}	57	1-1/2	7,968	10×10^{-5}	83	2-3/8	8,198	2.9
CH_4	5×10^{-5}	107	3/4		10×10^{-5}	165	1-1/4		
OF_2	5×10^{-5}	45	1-1/8	7,874	10×10^{-5}	69	1-1/2	7,921	0.6
CH_4	5×10^{-5}	105	3/4		10×10^{-5}	165	1-1/4		
F_2	5×10^{-5}	59	1-3/8	7,993	10×10^{-5}	78	2-1/2	8,078	0.6
NH_3	10×10^{-5}	16	*		20×10^{-5}	18	*		

*MINIMUM VALUE 1/4-IN. FOAM.

Table 30

MARS ORBITER SENSITIVITY TO PROPELLANT INITIAL CONDITION AND VENTING
(Sun-On-Tank, Pump-Fed, 205-Day Mission, $\alpha/\epsilon = 0.05/0.80$)

PROPELLANT	NONVENTED									VENTED			
	SATURATED H_2			TRIPLE-POINT H_2			50% SLUSH H_2			SATURATED H_2			
	P	T	WT	P	T	WT	P	T	WT	P	T	WT	BOILOFF (LB)
F_2	40	1-1/8	7238	40	1-1/8	7126	40	1-1/8	7067	15	1/2	6898	100
H_2	130	4-5/8	(REF)	95	3-3/4	(-1.5%)	82	3-1/2	(-2.4%)	76	2	(-4.7%)	
O_2	58	3/4	8477	58	3/4	8226	58	3/4	8141	15	1/2	8011	140
H_2	96	4-5/8	(REF)	66	3-3/4	(-3%)	66	3	(-4.0%)	64	2	(-5.5%)	

P = TANK MAXIMUM OPERATING PRESSURE (PSIA)
T = INSULATION THICKNESS (IN.)
WT = PROPULSION MODULE WEIGHT (LB)

Table 31

MARS ORBITER SENSITIVITY TO ORIENTATION
(Pump-Fed, Nonvented, 205-Day Mission, Optimum α/ϵ)

Propellant	Sun on Tank			Sun on Capsule			Weight Change (%)
	P[a]	T[b]	WT[c]	P	T	WT	
F_2 H_2	40 130	1-1/8 4-5/8	7,238	<15 80	Min 1-3/4	6,766	-7.0
O_2 H_2	58 96	3/4 4-5/8	8,477	<15 72	Min 1-3/4	7,839	-8.1
FLOX CH_4	57 107	1-1/2 3/4	7,968	32 <15	Min Min	7,752	-2.8
OF_2 CH_4	45 107	1-1/8 3/4	7,874	<15 <15	Min Min	7,707	-2.2
F_2 NH_3	59 16	1-3/8 Min	7,993	<15 <15	Min Min	7,843	-1.9
N_2O_4 A-50	<15 <15	Min Min	9,535	<15 <15	2 2	9,744	+2.2
ClF_5 MHF-5	<15 <15	Min Min	9,220	<15 <15	3/8 3/8	9,244	+0.3

(a) Tank maximum operating pressure (psia)
(b) Insulation thickness (in.)
(c) Propulsion module weight (lb)

reduced from 4-5/8 to 1-3/4 in., and the system weight for the F_2/H_2 propulsion module reduced from 7,238 to 6,766 lb by orienting the vehicle so that the propellant tanks are shaded. This effect is true for all of the cryogens and space-storable propellants. For the earth storables, a sun-orientation is more advantageous because a sun-shielded orientation requires thicker insulation to prevent the propellants from freezing.

6.8 ENGINE DESIGN VARIABLES

In addition to selecting the optimum vehicle orientation and surface coating, engine design variables can be optimized for each vehicle. An analysis of this type was conducted for the Mars Orbiter pump-fed systems. Variations in propellant mixture ratio, nozzle expansion ratio, and chamber pressure were considered. The specific impulse and engine weight are given in Table 32 of these variables. The most significant effect is obtained by varying the nozzle expansion ratio. The weight penalty is very small compared to the increase in specific impulse. The only limitation appears to be the vehicle volumetric constraint.

The vehicle system weights were determined for all propellant combinations. These data are presented in Figs. 35 through 40. The system weights reaffirm the raw-data conclusions. The optimum values were identical to the baseline design points except for the F_2/H_2 mixture ratio, in which a mixture ratio of 14 resulted in a lower weight, and the F_2/NH_3 chamber pressure variation, which had an engine data discontinuity because of the large chamber-pressure range. In conclusion, the nominal operating conditions selected are near optimum. The expansion ratio should be maximized within the volumetric constraints imposed by the system.

Table 32

DESIGN VARIABLES FOR 8000-LB-THRUST PUMP-FED ENGINES

		Mixture Ratio Variation $\epsilon = 100$				Expansion Ratio Variation $\epsilon = 40$	$\epsilon = 60$	$\epsilon = 100$	$\epsilon = 150$	Chamber Pressure Variation $\epsilon = 100$			
F_2/H_2	MR	11	12	13*	14	13	13	13*	13	13	13	13*	
	P_C (PSI)	900	900	900*	900	900	900	900*	900	600	750	900*	
	I_{SP} (SEC)	468.9	468.6	468.0*	467.0	456.4	462.2	468.0*	471.9	464.4	466.3	468.0*	
	WT (LB)	152	152	152*	152	152	152	152*	154	145	145	152*	
O_2/H_2	MR	5.5	5.75	6.0*	6.25	6.0	6.0	6.0*	6.0	6.0	6.0	6.0*	
	P_C	900	900	900*	900	900	900	900*	900	600	750	900*	
	I_{SP}	452.9	452.1	451.0*	449.6	439.4	443.7	451.0*	456.4	449.5	450.6	451.0*	
	WT	173+	173+	173+*	173+	152	152	173+*	182+	166+	166+	173+*	
FLOX/ CH_4	MR	5.25	5.50	5.75*	6.0	5.75	5.75	5.75*	5.75	5.75	5.75*		
	P_C	600	600	600*	600	600	600	600	600	300	600*		
	I_{SP}	408.0	409.2	410.0*	405.2	399.5	404.0	410.0*	413.7	405.4	410.0*		
	WT	152	152	152*	152	151	151	152*	156	145	152*		
OF_2/ CH_4	MR	4.7	5.0	5.3*	5.6	5.3	5.3	5.3*	5.3	5.3	5.3*		
	P_C	600	600	600*	600	600	600	600*	600	300	600*		
	I_{SP}	409.8	410.5	410.0*	409.3	396.8	402.8	410.0*	414.8	405.4	410.0*		
	WT	152	152	152*	152	151	151	152*	156	145	152*		
F_2/NH_3	M_R	2.9	3.1	3.3*	3.5	3.3	3.3	3.3*	3.3	3.3	3.3	3.3*	
	P_C	1500	1500	1500*	1500	1500	1500	1500*	1500	500	1000	1500*	
	I_{SP}	407.3	407.8	408.0*	402.3	399.0	403.3	408.0*	411.3	402.7	406.6	408.0*	
	WT	192	192	192*	192	191	191	192*	194	152	152	192*	
N_2O_4 A-50	MR	1.6	1.8	2.0*	2.2	2.0	2.0	2.0*	2.0	2.0	2.0	2.0*	2.0
	P_C	750	750	750*	750	750	750	750	750	350	550	750*	950
	I_{SP}	325.1	332.2	335.0*	333.5	325.3	330.1	335.0*	338.2	333.1	334.5	335.0*	335.2
	WT	158	158	158*	158	158	158	158*	162	152	158	158*	165

* Baseline system.
+ Two position nozzle.

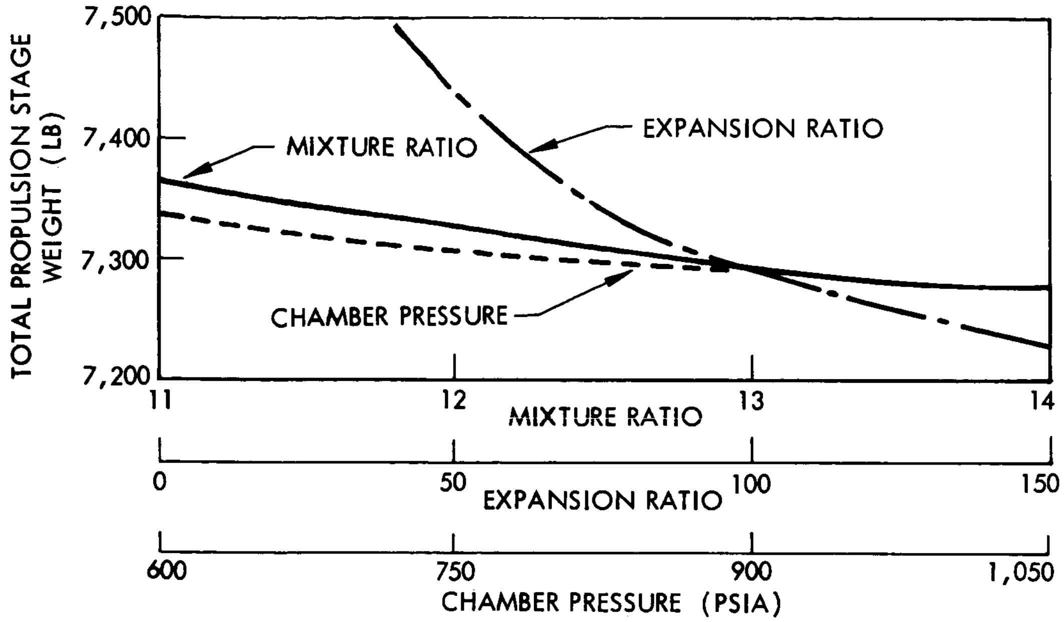


Fig. 35 Engine Design Parameter Sensitivity for F_2/H_2 Propellant

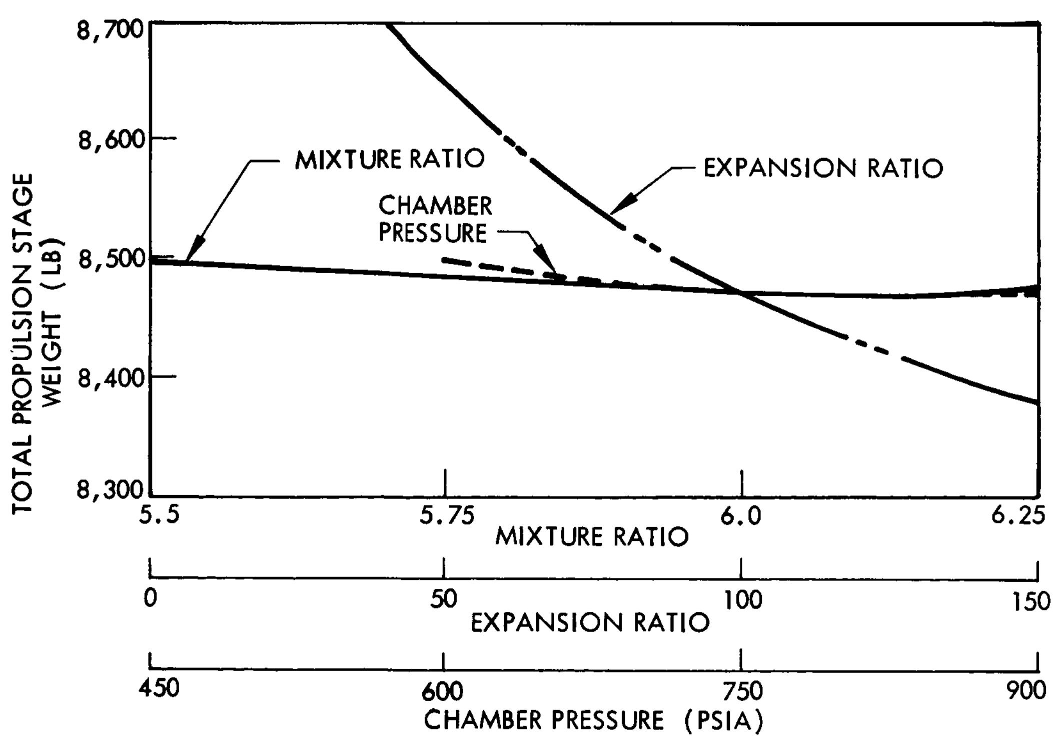


Fig. 36 Engine Design Parameter Sensitivity for O_2/H_2 Propellant

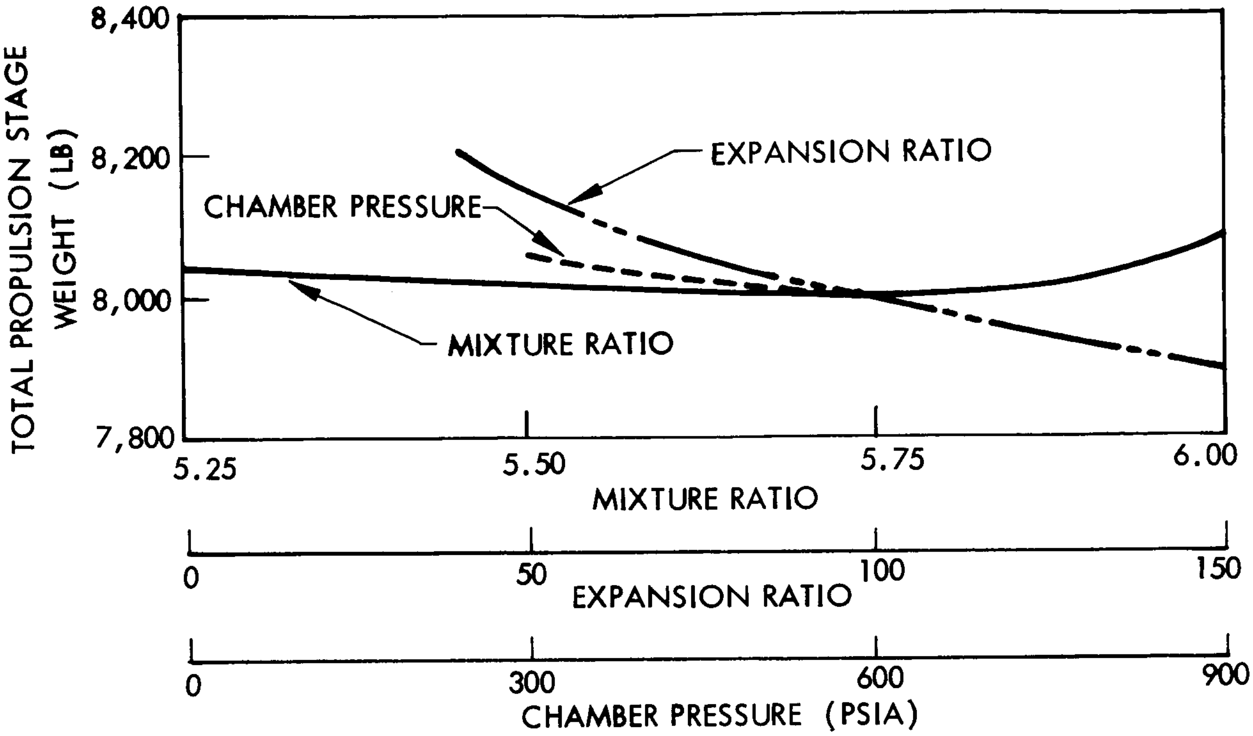


Fig. 37 Engine Design Parameter Sensitivity for FLOX/CH_4 Propellant

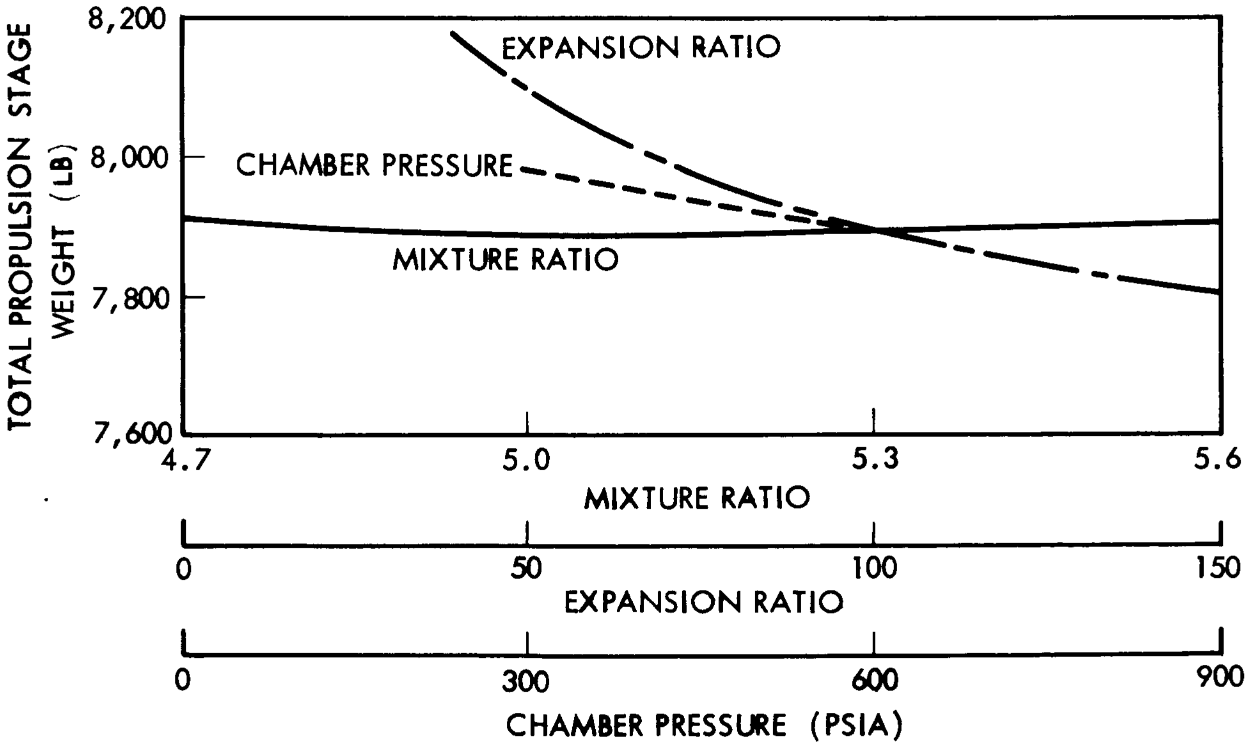


Fig. 38 Engine Design Parameter Sensitivity for OF_2/CH_4 Propellant

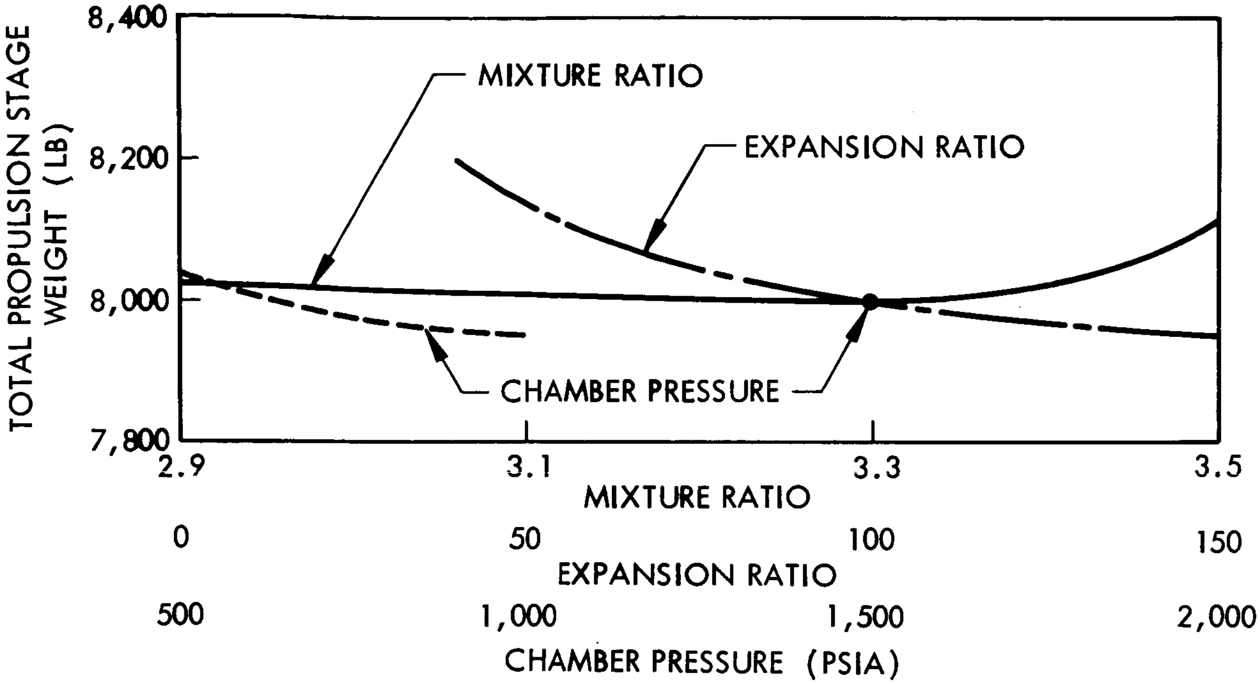


Fig. 39 Engine Design Parameter Sensitivity for F_2/NH_3 Propellant

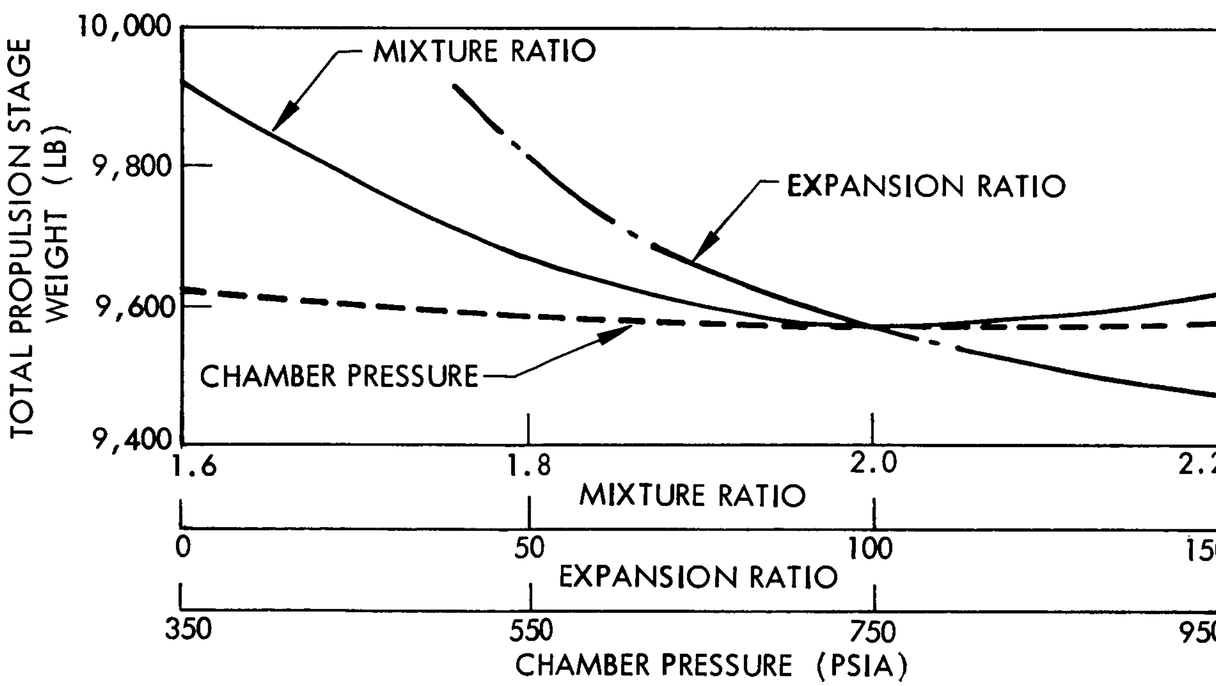


Fig. 40 Engine Design Parameter Sensitivity for N_2O_4/A-50 Propellant

6.9 WORST-ON-WORST ANALYSIS

To summarize the sensitivity analysis, a combination of adverse design conditions were considered. In this case the insulation conductivity was doubled, the heat leaks were doubled, only white paint surfaces could be used to obtain the best α/ϵ values, and the helium pressurization tanks were to be man rated. To evaluate the Mars Orbiter worst-on-worst requirements, the following specific conditions were analyzed:

- Vehicle Orientation
  - Sun on capsule for all propellants except earth storables
  - Sun on tanks for N_2O_4/A-50 and ClF_5/MHF-5 using $\alpha/\epsilon = 0.6/0.91$ and $\alpha/\epsilon = 0.3/0.95$
  - Sun on tanks and capsule for F_2/NH_3 using $\alpha/\epsilon = 0.3/0.95$
- High insulation conductivity (values of two times the baseline)
  - $k = 5.0 \times 10^{-5}$ Btu/hr-ft-°R for H_2
  - $k = 10.0 \times 10^{-5}$ Btu/hr-ft-°R for O_2, F_2, FLOX, CH_4, and OF_2
  - $k = 20.0 \times 10^{-5}$ Btu/hr-ft-°R for NH_3, N_2O_4, A-50, ClF_5, and MHF-5
- Double the heat leaks (values of two times the baseline)
  - Half the penetration (propellant feed and pressurant lines) thermal resistance
  - Half the support strut thermal resistance

The F_2/NH_3 propellant combination was analyzed for both the sun on capsule condition and sun on tanks with an α/ϵ of 0.30/0.95 (white paint) to determine the optimum orientation. Also, the sun on tank cases were all analyzed with an α/ϵ of 0.3/0.95 and an α/ϵ of 0.3/0.95 and an α/ϵ of 0.6/0.91 to determine the optimum surface finish.

For the F_2/NH_3 propellant combination, the sun on capsule condition resulted in the minimum system weight and is, therefore, the only one presented. This occurs even with an insulation thickness of 2.5 in. for NH_3 in order to prevent freezing because, with the sun on the tanks, the F_2 requires over 3 in. of insulation, and the F_2 tank is slightly larger than the NH_3 tank. The N_2O_4/A-50 propellant combination results in minimum system weight with an α/ϵ of 0.6/0.91, whereas, the ClF_5/MHF-5 combination optimizes with an α/ϵ of 0.3/0.95 because of the lower freezing points.

Many of the propellants experience a net heat loss during the mission; therefore, only minimum insulation thicknesses are required. The propulsion module weights are presented in Table 33. Also included in the table are the required operating pressure and insulation thickness. The weight increases for the worst-on-worst condition are low for all of the propellants, varying from 0.6 percent for ClF_5/MHF-5 to 1.5 percent for F_2/NH_3.

Table 33

MARS ORBITER WORST-ON-WORST ANALYSIS

Propellant	Baseline			Worst-On-Worst				Orientation
	Operating Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	Operating Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	Weight Increase (%)	
F_2 H_2	< 15 80	Min[(a)] 1-3/4	6,766	18 59	1/2 2	6,839	0.8	Sun on capsule
O_2 H_2	< 15 72	Min 1-3/4	7,839	18 52	1/2 2	7,930	1.4	Sun on capsule
FLOX CH_4	32 < 15	Min Min	7,752	52 22	1/2 1/2	7,845	1.2	Sun on capsule
OF_2 CH_4	< 15 < 15	Min Min	7,707	19 22	1/2 1/2	7,790	1.1	Sun on capsule
F_2 NH_3	< 15 < 15	Min Min	7,843	52 18	1/2 2-1/2	7,964	1.5	Sun on capsule
N_2O_4 A-50	< 15 < 15	Min Min	9,535	25 15	5/8 5/8	9,641	1.7	Sun on tank
ClF_5 MHF-5	< 15 < 15	Min Min	9,220	20 15	3/8 3/8	9,278	0.6	Sun on tank

(a) Min represents less than 1/2-in. of multilayer insulation required.

Section 7
REFERENCES

1. Scaling Laws and Standard Models provided by NASA for use in Contract NASw-1644. All information in this reference was extracted from Ref. 2.

2. Deerwester, T. M. and Norman, S. M., "Reference System Characteristics for Manned Stopover Missions to Mars and Venus," OART – Mission Analysis Division Paper, MS-67-4, Jul 10, 1967

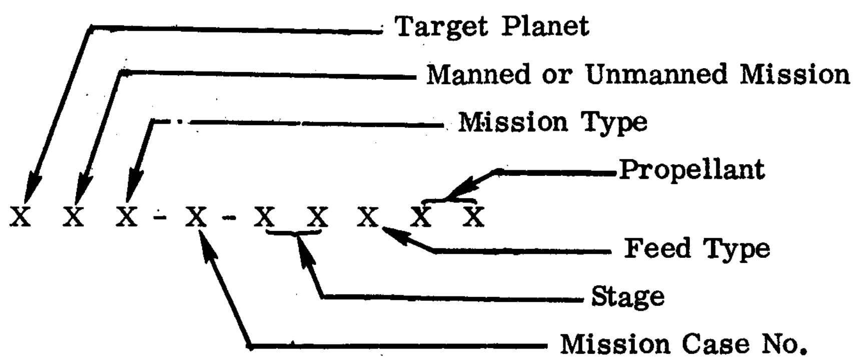
3. Cosby, W. A. and Lyle, R. G., "Interplanetary Meteoroid Shielding Requirements," Memorandum to Mission Analysis Division, Aug 15, 1966

4. -----, "The Meteoroid Environment and Its Effects on Materials and Equipment," NASA SP-78, 1965

Appendix A
IDENTIFICATION CODE AND MISSION DESCRIPTIONS

A.1 IDENTIFICATION CODE

An eleven-character identification code is described as follows, and specified for use during the Propellant Selection Study.



A.1.1 Mission Code

First Character – Target Planet

H = Mercury
V = Venus
E = Earth
M = Mars
J = Jupiter
S = Saturn
L = Moon of Earth

Second Character – Manned or Unmanned

M = Manned
U = Unmanned

Third Character — Mission Type

F	=	Flyby
L	=	Lander
S	=	Surface Station
O	=	Orbiter
P	=	Probe

Fourth Character — A dash for separation

Fifth Character — A number designating a particular mission time and trajectory

A.1.2 Spacecraft Stage Code

This follows the mission code and identifies each stage.

Sixth Character — A dash for separation from the mission code

Seventh and Eighth Characters — Stage type identification

ED	=	Earth Departure,
Ml	=	Midcourse, 1, 2, 3, etc., for sequence of midcourse and orbit correction stages
OI	=	Orbit Injection
DS	=	Descent Stage
AS	=	Ascent Stage
PD	=	Planet Departure (from orbit for planets; from lunar surface for the moon)

A.1.3 Propellant Feed System Type Code

Ninth Character

+	=	Pump Fed
-	=	Pressure Fed

A.1.4 Propellant Combination Code

Tenth and Eleventh Characters

01	=	$F_2 - H_2$
02	=	$O_2 - H_2$
03	=	H_2O_4 - A50

04 = OF_2 - CH_4
05 = OF_2 - C_3H_8
06 = OF_2 - B_2H_6
07 = OF_2 - MMH
08 = FLOX - CH_4 (82.5% F_2/17.5% O_2)
09 = FLOX - C_3H_8 (76% F_2/24% O_2)
10 = F_2 - NH_3
11 = $C\ell F_5$ - MHF-5

A.1.5 Examples

Example No. 1:

MML-1-ED+O1

Mars Manned Lander (and Orbiter), - Case No. 1, - Earth Departure Stage with pump fed F_2 - H_2 propulsion system.

Example No. 2:

LMS-2-PD-06

Lunar Manned Return Mission, - Case No. 2, - Planet departure stage (direct from Lunar surface), pressure fed OF_2 - B_2H_6 propulsion system

A.2 MISSION DESCRIPTIONS

A.2.1 Unmanned Mission Descriptions (See Table A-1)

MUF-1: Mariner-Mars Flyby/Probe – 1971. Objective: To obtain final support data on the surface and atmosphere of Mars in preparation for Voyager orbiter/lander missions. The spacecraft consists of an 800-lb bus which would fly past Mars and drop a 100-lb probe. This spacecraft would be launched by an Atlas-Centaur launch vehicle. No specific documentation is available for this mission except Ref. A-1.

Table A-1

MISSION REQUIREMENTS INVENTORY — UNMANNED MISSIONS

Mission Name and Code Number	Stage Code No.	Mission Year	Payload (lbm)	Stage Diameter (ft)	Δ V (ft/sec)	Exposure Time (days)	Thrust Nominal (lbt)	No. of Engine Starts	Class of Booster	References
MUF-1 (Mariner-Mars)	M1 (Midcourse)	1971	800 +100 Probe	10.0	330	190	50	3	Atlas Centaur	A-1
VUF-1 (Mariner-Venus)	M1 (Midcourse)	1975	800 +300 Probe	10.0	330	176	50	3	Atlas Centaur	A-1
MUO-1 Voyager-Mars	OI (Orbit Injection)	1973	8,143	21.6	6,950	195	8,000	6	S-V	A-1, A-2
	DS (Lander)	1973	5,000	21.6	400 to 1,000	195	4,000	1	—	—
MUO-2 (Voyager-ABL)	OI (Orbit Injection)	1977	13,500	21.6	5,000	325	8,000	6	S-V	A-1
	DS (Lander)	1977	10,000	21.6	400 to 1,000	325	8,000	1	—	—
VUO-1 (Voyager-Venus	OI (Orbit Injection)	1977	4,500 +2,500 Probe	10.0 to 21.6	13,500	140	8,000	6	260/ S-IVB Centaur	A-1
JUO-1 (Jupiter Orbiter)	OI (Orbit Injection)	1981	2,000	10.0	7,600	650	2,000	6	260/ S-IVB Centaur	A-1
SUO-1 (Saturn Orbiter)	OI (Orbit Injection)	1984	2,000	10.0	6,000	1,450	2,000	6	260/ S-IVB Centaur	A-1

Propulsion requirements for the bus consist only of a 100-meter/sec midcourse correction (two burns).

VUF-1: Mariner-Venus Flyby/Probe – 1975. Objective: To obtain data on the atmosphere and environment of Venus in preparation for later voyager orbiter missions with probes. The spacecraft consist of an 800-lb bus which would fly past Venus and drop a 100-lb probe. This spacecraft would be launched by an Atlas-Centaur launch vehicle. No specific documentation is available for this mission except Ref. A-1. Propulsion requirements for the bus consist only of a 100-meter/sec midcourse correction.

MUO-1: Voyager-Mars-Orbiter/Lander – 1973. Objective: To obtain information relevant to the existence and nature of extraterrestrial life; the atmosphereric, surface, and body characteristics of Mars; and the planetary environment by performing unmanned experiments in the orbit about and on the surface of Mars. The spacecraft consists of an ~ 3,000-lb bus which would orbit the planet, a 5,000-lb lander, and propulsion system. These spacecraft are sized for launch in tandem (two spacecraft) on one Saturn V. The most current documentation consists of Refs. A-2 and A-3 and "Summary of the Voyager Program," NASA/OSSA, Jan 1967. Propulsion systems that have been considered for this application are as follows:

- Solid propellant motor for orbit insertion plus liquid system for midcourse
- LEM descent propulsion system
- Transtage propulsion system (modified)

Propulsive requirements for the bus are 200 meters/sec (two burns) for midcourse; 2.0 km/sec for orbit insertion; and 100 meters/sec for orbit trim. Total velocity requirements are normalized to 6,950 ft/sec. Capsule decleration is assumed to be a combination of aerodynamic drag, parachute, and retropropulsion. Velocity at ignition of retrosystem would be between 400 and 1,000 ft/sec at an altitude of 10,000 to 20,000 ft. One of the propulsion systems considered for this application is an N_2O_4/ UDMH engine.

MUO-2: Voyager – Mars Orbiter/ABL – 1977. Objective: To obtain information relevant to the existence and nature of extraterrestrial life; the atmospheric, surface, and body characteristics of Mars; and the planetary environment by performing unmanned experiments in orbit about and on the surface of Mars with greater emphasis on life detection lander experiments than the earlier orbiter/lander mission. The spacecraft consists of a 3,500-lb bus which would orbit the planet, a 10,000-lb lander, and propulsion system. These spacecraft are sized for launch in tandem (two spacecraft) on one Saturn V. The most current documentation consists of Refs. A-1 and A-2.

Propulsion Systems that have been considered for this application are as follows:

- Solid propellant motor for orbit insertion plus liquid system for midcourse
- LEM descent propulsion system
- Transtage propulsion system (modified)

Propulsive requirements for the bus are 200 meters/sec (two burns) for midcourse, maximum possible for orbit insertion within spacecraft weight constraints (~ 4,000 f5/sec), and 100 meters/sec for orbit trim. Capsule deceleration is assumed to be a combination of aerodynamic drag, parachute, and retropropulsion. Velocity at ignition of retrosystem would be between 400 and 1,000 ft/sec.

VUO-1: Voyager – Venus Orbiter/Probe – 1977. Objective: To obtain detailed information about the atmosphere of Venus, including composition, temperature, pressure, and density profiles, and to assess the shape of the planet and its particle field environment. The spacecraft consists of a 2,000-lb bus which would orbit the planet, a 2,500-lb probe, and a propulsion system. This spacecraft is sized for launch by a 260/S-IVB Centaur launch vehicle. No specific documentation is available for the Voyager-Venus Mission except Ref. A-1. Propulsive requirements for the bus are 200 meters/sec (two burns) for midcourse, 12,500 ft/sec for orbit insertion, and 100 meters/sec for orbit trim.

JUO-1: Voyager Advanced Planetary – Jupiter Orbiter – 1981. Objective: To obtain initial detailed data on the atmosphere of Jupiter and to define the shape and strength of its magnetic fields and its surrounding environment. The spacecraft consists of a

2,000-lb bus which would orbit the planet, and a propulsion system. This spacecraft is sized for launch by a 260/S-IVB/Centaur launch vehicle. No specific documentation is available for the Jupiter Orbiter mission except Ref. A-1. Propulsive requirements for the bus are midcourse – 200 meters/sec (two burns) for midcourse, 6,600 ft/sec for orbit insertion, and 100 meters/sec for orbit trim.

SUO-1: Voyager Advanced Planetary – Saturn Orbiter – 1984. Objective: To obtain initial detailed data on the magnetic fields, atmosphere, mass, environment, and rings of Saturn. The spacecraft consists of a 2,000-lb bus which would orbit the planet, and a propulsion system. This spacecraft is sized for launch by a 260/S-IVB/Centaur launch vehicle. No specific documentation is available for the Saturn Orbiter mission except Ref. A-1. Propulsion requirements for the bus 200 meters/sec (two burns) for midcourse, 5,000 ft/sec for orbit insertion, and 100 meters/sec for orbit trim.

A.2.2 Manned Mission Descriptions (See Table A-2)

LMS-1-PD: Lunar Manned Station – Personnel Department Stage Direct from Lunar Surface. Objective: To provide the propulsion functions necessary to return six men direct from the lunar surface to an aerodynamic entry trajectory at earth. The liftoff configuration consists of a six-man command module mated to a propulsion stage. The propulsive stage includes attitude control, electrical power, and environmental control support functions, as well as primary propulsion and midcourse corrections. The most current information is Ref. A-3. The propulsive stage must provide a ΔV of 9. 9,186 ft/sec for a payload of 19,340 lb after having been in dorman storage on the lunar surface for 178 days.

MMF-1 and VMF-1: Manned Planetary Flyby Missions – Mars and Venus. Objective: To conduct the earliest possible manned interplanetary mission to the proximity of the near planets, Mars and Venus, and to provide significant scientific and engineering knowledge about these planets. The spacecraft could be launched using a variety of injection stages, and would have a payload that varies with planet and opportunity. The spacecraft consists primarily of a manned mission module, an earth re-entry module,

Table A-2

MISSION REQUIREMENTS INVENTORY — MANNED MISSIONS

Mission Name and Code Number	Stage Code No.	Mission Year	Payload (lbm)	Stage Diameter (ft)	ΔV (ft/sec)	Exposure Time (days)	Thrust Nominal (lbt)	No. of Engine Starts	Class of Booster	References
LMS-1 (Lunar Manned Surface Station)	PD (Planet Departure)	1978	19,340	21.6	9,186	178	15,000	3	Uprated Saturn	A-3
MMF-1 (Mars Manned Flyby)	ED (Earth Departure)	1977	224,000	21.6	7,340*	5	30,000	1	Uprated Saturn (Multiple Launch)	A-4
(+ Probes)	M1 (Midcourse)	1977	224,000	<21.6	660	695	1,000	2		
	Probe 1 (Orbiter)	1977	1,000 (28,000 Gross)	<21.6	21,000	150	4,000	2		
	Probe 2 (Soft Lander)	1977	4,000 (5,000 Gross)	<21.6	1,000	150	8,000	2		
	Probe 3 (MSSR)	1977	<100 (3,800 Ascent Weight)	<21.6	36,000	150	+			A-5
VMF-1 (Venus Manned Flyby)	ED (Earth Departure)	1977	135,000	21.6	2,880*	376	30,000	1	Uprated Saturn	A-4
(+ Probes)	M1 (Midcourse)	1977	135,000	<21.6	660	376	1,000	2		

*Abort Δ V, dependent on mission mode, is not included.
\+ To be determined.. This is a multi-stage system.

Table A-2 (Cont.)

Mission Name and Code Number	Stage Code No.	Mission Year	Payload (lbm)	Stage Diameter (ft)	ΔV (ft/sec)	Exposure Time (days)	Thrust Nominal (lbt)	No. of Engine Starts	Class of Booster	Refer- ences
VMF-1 (Cont'd) (Venus Manned Flyby) (+ Probes)	Probe 1 (Orbiter)	1977	1,500 (9,000 Gross)	<21.6	13,000	115	4,000	2		—
MML-1 (1982-Direct 30-Day Stay 420-Day Duration)	ED	1982	660,000	33.0	12,900	60***	100K per Module	1	Im- proved Saturns	A-6
	M1	1982	655,000	33.0	250	191	2,000	2		
	OI	1982	650,000	33.0	500	191	2,000	2		
	DS	1982	140,000 (Gross)**	<33.0	2,000	221	100K	2		
	AS	1982	80,000 (Gross)**	<33.0	15,500	221	50,000	4 min.		
	PD	1982	92,000	33.0	15,000	221	100K per Module	1		
	M2	1982	92,000	33.0	250	450	1,000	2		
MML-2 (1982-Swingin 30-Day Stay 540-Day Duration)	ED	1982	770,000	33.0	12,700	60	100K per Module	1	Im- proved Saturns	A-6
	M1	1982	765,000	33.0	250	250	2,000	2		
	OI	1982	760,000	33.0	500	250	2,000	2		

** At start of burn.

*** Assume 60 and 120 days to check sensitivity.

Table A-2 (Cont.)

Mission Name and Code Number	Stage Code No.	Mission Year	Payload (lbm)	Stage Diameter (ft)	ΔV (ft/sec)	Exposure Time (days)	Thrust Nominal (lbt)	No. of Engine Starts	Class of Booster	References
MML-2(Cont'd) (1982-Swingin 30-Day Stay 540-Day Duration)	DS	1982	140,000 (Gross)*	<33.0	2,000	280	100K	2		—
	AS	1982	8,000 (Gross)*	<33.0	15,000	280	50,000	4		
	PD	1982	92,000	33.0	16,000	280	100K per Module	1		
	M2	1982	92,000	33.0	500	540		4		
MML-3 (1982-Swingin 100-Day Stay 540-Day Duration)	ED	1982	785,000	33.0	12,900	60**	100K per Module	1	Im-proved Saturns	A-6
	M1	1982	780,000	33.0	250	200	2,000	2		
	OI	1982	775,000	33.0	500	200	2,000	2		
	DS	1982	150,000 (Gross)*	<33.0	2,000	300	100K	2		
	AS	1982	80,000 (Gross)*	<33.0	15,500	300	50,000	4		
	PD	1982	92,000	33.0	16,000	300	100K per Module	1		
	M2	1982	92,000	33.0	500	540	1,000	4		

*At start of burn.
** Assume 60 and 120 days to check sensitivity.

Table A-2 (Cont.)

Mission Name / Code Number	Stage Code No.	Mission Year	Payload (lbm)	Stage Diameter (ft)	Δ V (ft/sec)	Exposure Time (days)	Thrust Nominal (lbt)	No. of Engine Starts	Class of Booster	References
VMO-1 (1985-Direct 30-Day Stay 440-Day Duration Aero-Braking)	ED	1985	444,000	33.0	11,600	60*	100K per Module	1	Im-proved Saturns	A-6
	M1	1985	440,000	33.0	250	143	2,000	2		
	OI	1985	436,000	33.0	500	143	2,000	2		
	PD	1985	92,000	33.0	14,000	173	100K per Module	1		
	M2	1985	92,000	33.0	250	470	1,000	2		
MMO-1 (1980-Direct 30-Day Stay 430-Day Duration Direct)	ED	1980	634,000	33.0	13,700	60*	100K per Module	1	Im-proved Saturns	A-6
	M1	1980	630,000	33.0	250	197	2,000	2		
	OI	1980	626,000	33.0	500	197	2,000	2		
	PD	1980	92,000	33.0	15,700	227	100K per Module	1		
	M2	1980	92,000	33.0	250	460	1,000	2		
EMO-1 (Earth Manned Orbiter)	OI	1973	95,400	21.6	{1,600 / 8,064 / 9,570} 19,234	0 / 0.05 / 0.3	200K	3	Uprated Saturn	A-7
	DS	1973	13,000	12.8	{2,100 / 7,650} 9,750	60 / 60	20,000	4		

*Assume 60 and 120 days to check sensitivity.

unmanned probes, and the corresponding propulsion systems. The documentation on which the mission requirements were based are Refs. A-4 and A-5.

MML-1, MML-2, and MML-3: Mars Manned Lander Missions. Objective: To land men on Mars and to obtain scientific and engineering data. The six-man spacecraft consists of a manned mission module, a three-man Mars excursion module, and an earth reentry module. Propulsion stages include earth departure, midcourse, Mars orbit injection, MEM descent and ascent, and Mars departure. Atmospheric braking would be used for entry to orbit and landing at Mars for reentry on return to earth. Multiple launches of an improved Saturn class booster would be required to place segments of the system in earth orbit for assembly. MML-1 is a direct flight to Mars with a 30-day stay at the planet. MML-2 is a Venus swingby flight to Mars, with a 30-day stay at Mars. MML-3 is a Venus swingby flight to Mars with a 100-day stay at Mars.

Documentation used in deriving the mission requirements was primarily from Ref. A-6.

VMO-1 and MMO-1: Venus Manned Orbiter and Mars Manned Orbiter. Objective: To orbit men about Venus/Mars for 30 days and to obtain scientific and engineering data of Venus/Mars. The six-man spacecraft consists of a manned mission module and an earth reentry module. Propulsion stages include earth departure, midcourse, planet orbit injection, and planet departure. Atmospheric braking would be used for entry to orbit at the planet and for reentry on return to earth. Documentation used in deriving the mission requirements was primarily from Ref. A-6.

EMO-1: Earth Manned Orbiter. Objective: To station men in a synchronous orbit about earth for a period of 60 days to perform scientific and engineering experiments. This is a candidate AAP mission. The spacecraft are to be launched by two Saturn class boosters, rendezvous in synchronous orbit, and have a 60-day operational period. The first launch would consist of a command/service module and the second launch would consist of a modified LM and experiments. In the current mission planning, the SIV-B would burn three times. Injecting the payload into low orbit, ΔV_1 is 1,600 ft/sec;

injecting the payload into a higher orbit, ΔV_2 is 8,064 ft/sec; and circularizing the synchronous orbit, ΔV_3 is 5,970 ft/sec. The orbit phasing and descent propulsion would be accomplished using the SM engine. The phasing velocity requirement is 2,100 ft/sec and the descent is 7,650 ft/sec. The payload for the descent propulsion would be the CM with a mass of 13,000 lb. No specific documentation is available for this mission. Information was obtained through personal contact with members of the AAP Payload Integration Study team at Lockheed.

A.3 REFERENCES

A-1 OSSA PROSPECTUS 1966 — Appendix B

A-2 TRW Systems Group, "Voyager Support Study – LM Descent Stage Applications – Final Report," 04480-6008-R000, Contract JPL 951113, Feb 1967

A-3 Lockheed Missiles & Space Company, "Improved Lunar Personnel and Cargo Delivery System," Contract NAS8-21006

A-4 North American Aviation Space Division, "A Study of Manned Planetary Flyby Missions Using Saturn/Apollo Systems – Final Report," Contract NAS8-18025

A-5 J. M. Tschirgi, "MSSR Memorandum, Bellcom, Inc.," Jun 14, 1966

A-6 Martin Company, "Study of Spacecraft Propulsion Systems for Manned Mars and Venus Missions," ER 13919, Jul 1965

A-7 AAP mission using Saturn S-IVB stage for orbit transfer and injection. No documentation available.

Appendix B
VOYAGER BASELINE DESCRIPTION

B.1 INTRODUCTION

The Voyager spacecraft, as defined by TRW in Ref. A-3, has been chosen as one of the two reference spacecraft stages to be analyzed by Lockheed in Contract NASw-1644, "Propellant Selection for Spacecraft Propulsion Systems."

This document presents a summary description of the reference Voyager mission and spacecraft as extracted from Ref. A-3.

B.2 MISSION DESCRIPTION AND REQUIREMENTS

The Voyager spacecraft (Fig. B-1) must be capable of performing the following tasks on a mission to Mars:

- Maintain full-time, three-axis orientation as commanded (solar panels and propulsion module face the sun except during propulsion maneuvers)
- Communicate with Earth DSIF, accept commands
- Perform up to two midcourse trajectory corrections while enroute to Mars
- Perform an orbit insertion maneuver upon arrival at Mars
- Perform an orbit trim maneuver after the initial orbit has been established about Mars

Maneuvers, time from launch, thrust, and velocity increments provided by the spacecraft propulsion system are shown in Table B-1.

Table B-1

VOYAGER MANEUVER REQUIREMENTS

Maneuver	Time From Launch (days)	Thrust Level (lbf)	ΔV (ft/sec)
1st Midcourse	2	1,050	164
2nd Midcourse	165	1,050	164
Orbit Insertion	195	7,750	6,294
Orbit Trim	205	1,050	328
		Total	6,950 ft/sec

Accelerations experienced by the spacecraft during launch and maneuvers

Maneuver	Max Acceleration (Earth g's)	
Launch	Longitudinal	7.0
	Lateral	1.25
Orbit Insertion	Longitudinal	0.75[a]
Orbit Trim	Longitudinal	1.5[a]

(a) Assumes thrust = 8,000 lb

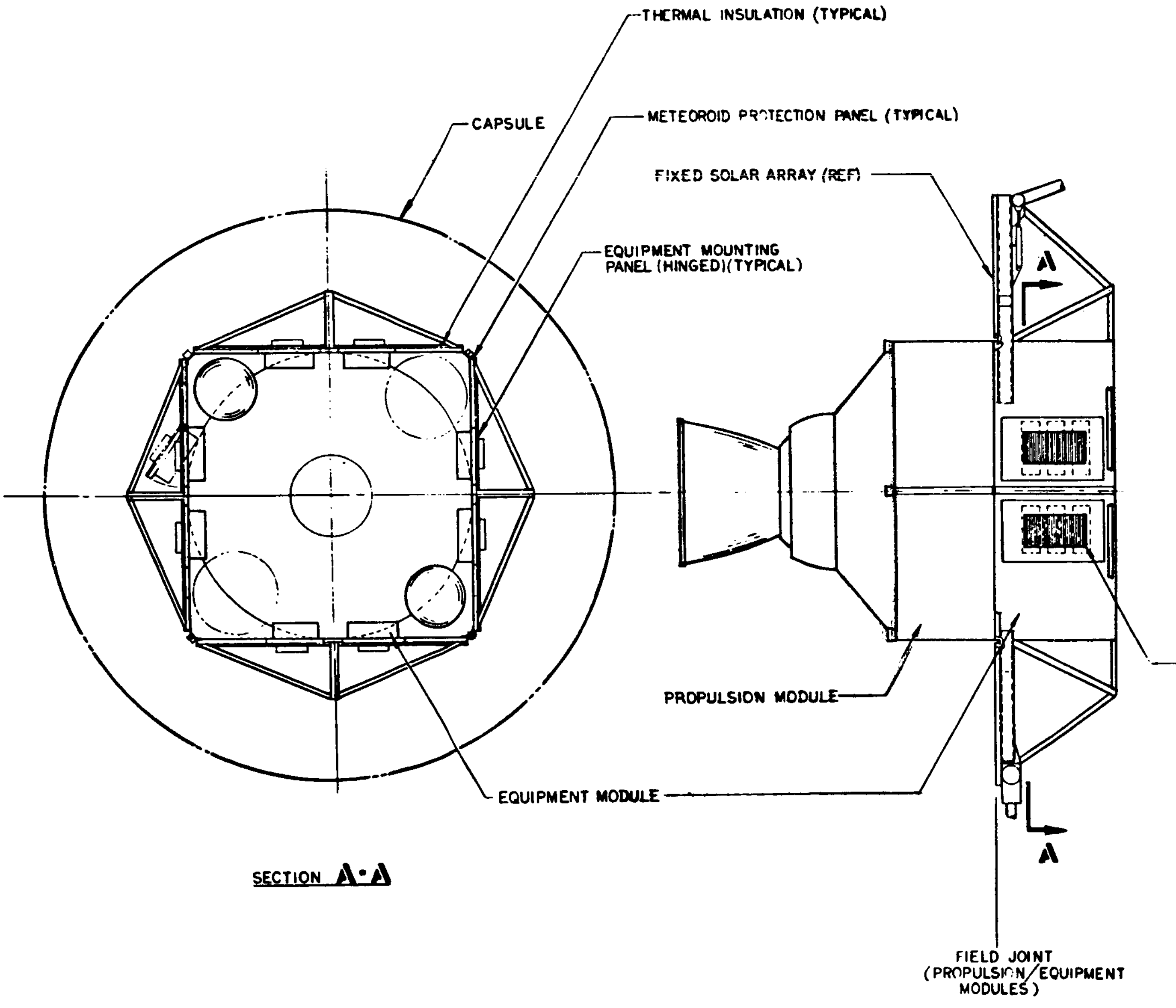
B.3 DESIGN CONSIDERATIONS/CRITERIA

B.3.1 General

The spacecraft (Fig. B-2) was configured based on the following general design considerations:

- Minimize spacecraft length
- Total spacecraft weight with adapter not to exceed 22,000 lb

FOLDOUT FRAME 1



SECTION A-A

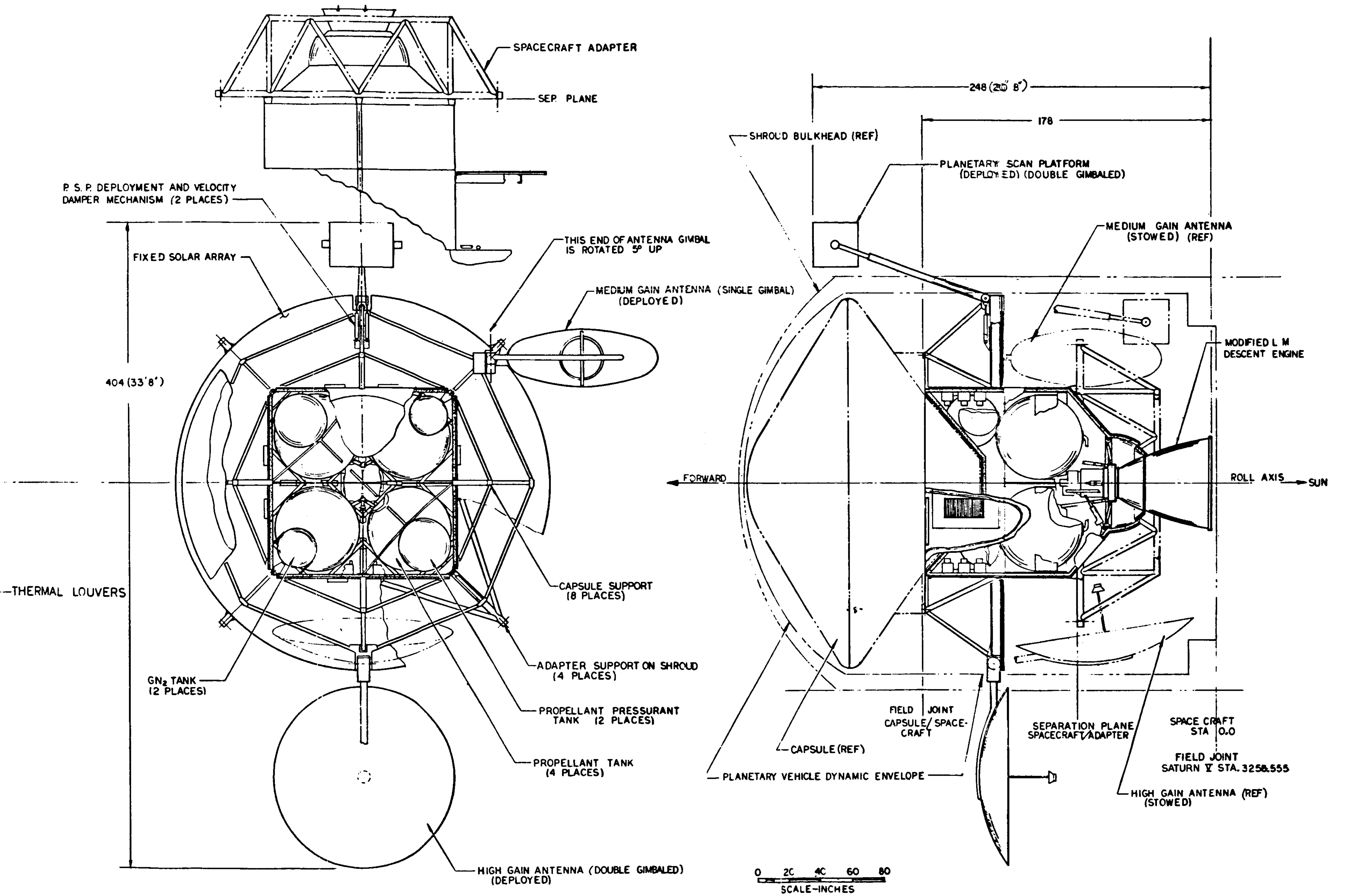


Fig. B-1 TRW Voyager Configuration

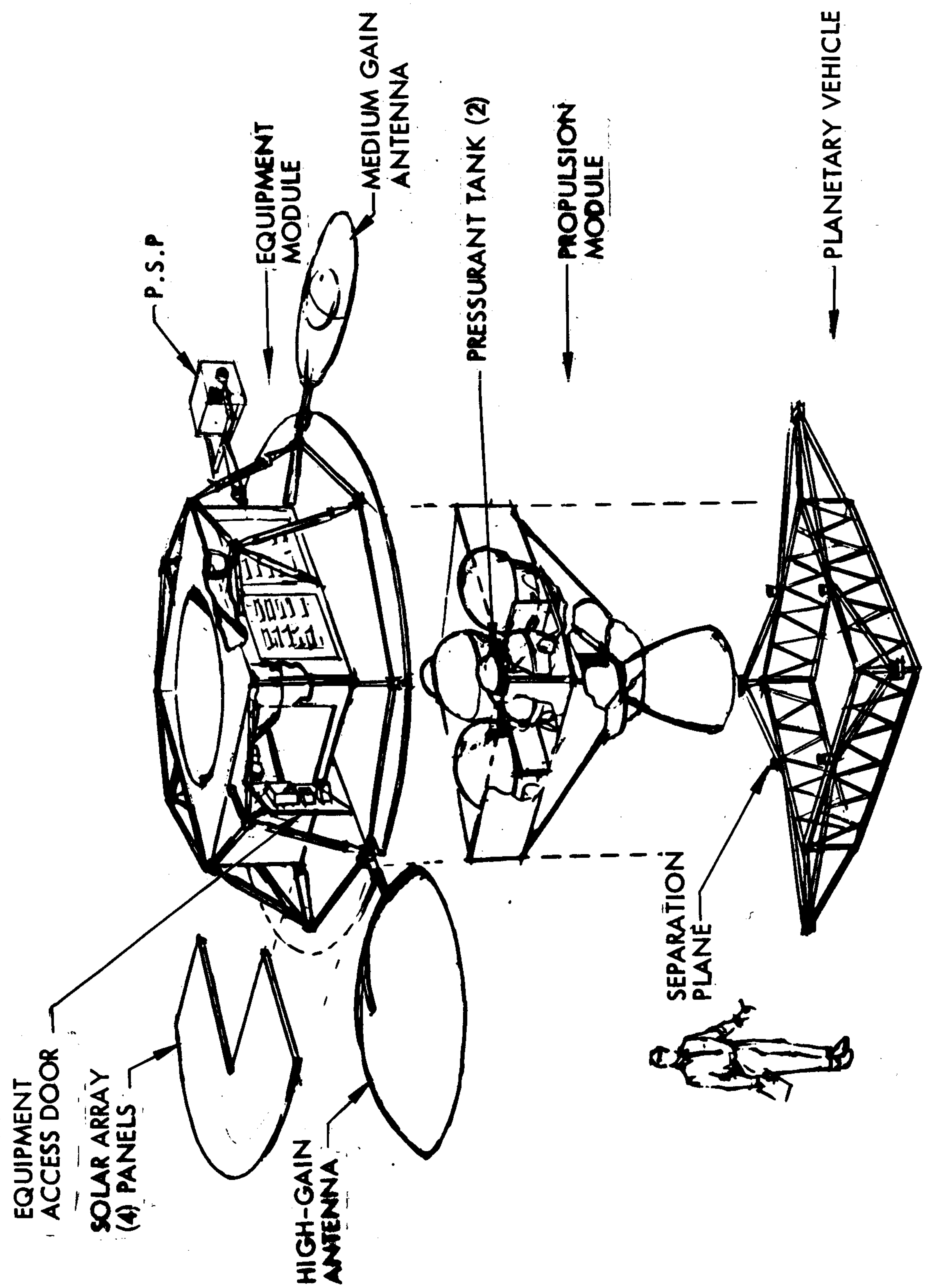


Fig. B-2 Modularity of the Voyager Spacecraft Configuration

- Up to 719 watts of power will be required from solar arrays (battery energy requirements are 1,270 w-hr maximum)
- Weight allocation for spacecraft mounted capsule support equipment is 50 lb
- The spacecraft shall be capable of completing its mission without separation of the capsule
- Minimizing loads and weight of spacecraft takes priority over spacecraft adapter weight
- Size propellant tanks for maximum allowable spacecraft weight of 22,000 lb to permit future growth without requiring tank redesign
- Separate equipment, propulsion, and capsule into modules to facilitate handling, testing, and field assembly

B.3.2 Propulsion System

The propulsion system design was based on use of the LM descent engine (modified) using N_2O_4/A-50 propellants. Four spherical tanks of equal size were used, resulting in an off-optimum mixture ratio (1.6 used versus 2.0 optimum). Tanks were assumed to be fabricated of titanium and mounted to the support structure through a 360-deg skirt arrangement. Propulsion system parameters are described in Table B-2.

Table B-2
PROPULSION SYSTEM CRITERIA

Area	Item	Criteria
Propellant	Oxidizer	N_2O_4 at 90.1 lb/ft^3 at 70°F
	Fuel	A-50 at 56.3 lb/ft^3 at 70°F
	Mixture Ratio (O/F)	1.6:1
	I_{sp}	285 sec at F = 1,050 lb 305 sec at F = 7,750 lb
	Residuals	382 + 0.0032 times usable propellant mass

Table B-2 (Cont.)

Area	Item	Criteria
Tanks	Type	Spherical
	Number	4
	Size	53-in. diameter
	Volume	$45.1\ ft^3 \times 4 = 180.4\ ft^3$
	Material	6Al - 4V titanium
	Ult. Stress Level	160,000 psi
	Tank Pressure	270 psia (max.)
	Factor of Safety	2.2
	Min. Skin Gage	0.020-in.
	Contingency	10 percent
	Ullage Volume	3 percent

B.3.3 Pressurization System

The propulsion system is pressure-fed. The pressurization system is described as follows:

Pressurization Schedule.

- Tanks pressurized to 235 psia with helium from a ground source prior to launch
- First midcourse correction uses available pressure in blowdown made (tank ullage pressure falls to 95 psi)
- Tanks are subsequently repressurized to 235 psia using helium from spacecraft pressurization system during and after each propulsion maneuver

The capsule is self contained except for power supplied from the equipment module. Capsule separation is accomplished after orbit insertion at Mars. The equipment module contains all spacecraft equipment except those items relating to the propulsion system. The propulsion module contains all items related to pressurization, propellant storage and delivery, and engine. The engine is gimbaled for thrust vector control. Propulsion module details are shown in Fig. B-3.

B.4.3 Weight Breakdown

A weight breakdown and spacecraft mass properties are presented in Tables B-3 through B-6. The allowable CG location is shown in Fig. B-4 and the spacecraft axis coordinate system is defined in Fig. B-5.

B.5 SUBSYSTEMS DESCRIPTION

Description of the spacecraft subsystems is limited here to those items influencing the propellant selection study configurations through physical arrangement, thermal effects, or structural protection.

B.5.1 Electrical Power

Primary power is supplied by fixed solar panels with a total area of 165 ft^2. Secondary power is provided by batteries mounted in the equipment module.

B.5.2 Meteoroid Protection

A meteoroid shield composed of an 0.020-in.-thick aluminum inside sheet, 2 in. of filler, and an 0.010-in.-thick aluminum outersheet encloses the equipment module and propulsion module. This design is based on the following ground rules:

- Tank wall = 0.030 in. (titanium)
- Meteoroid critical puncture mass = 0.0003 gram
- Meteoroid flux is based on near-earth observations and Mariner 4, data and is described in Ref. A-1
- Impact probability range = 0.0013 to 0.005/month

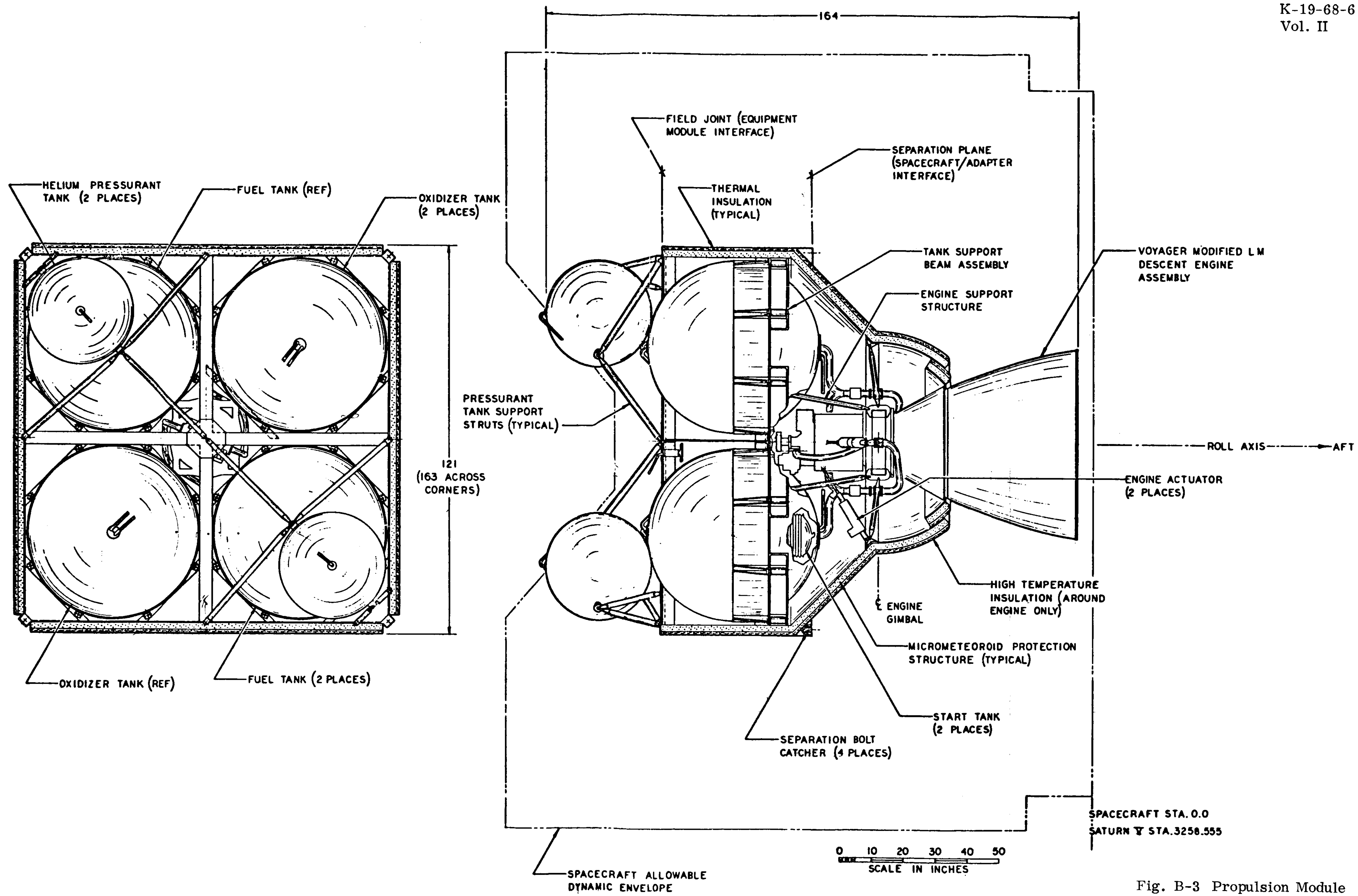


Fig. B-3 Propulsion Module

Table B-3

PLANETARY VEHICLE SUMMARY WEIGHT

Item	Weight (lb)
Flight Capsule	5,000.0
Flight Spacecraft Science Subsystems	400.0
Flight Spacecraft Capsule Bus Support Equipment	50.0
Flight Spacecraft Equipment Module	1,980.3
Structure	502.6
Thermal Control	132.2
Pyrotechnics	37.0
Power	364.1
Electrical Distribution	228.9
Guidance and Control	268.5
Communications	125.5
Telemetry and Command	90.5
Computing and Sequencing	36.0
Balance Weights	15.0
Contingency	180.0
Flight Spacecraft Propulsion Module	13,453.4
Structure	512.0
Thermal Control	29.4
Engine and Valves	426.5
Propellant Feed Assembly	363.1
Pressurization System	414.4
Contingency	174.5
Residuals (propellant and helium)	462.5
Usable Propellant	11,071.0
Planetary Vehicle Gross Weight	20,883.7
Planetary Vehicle Adapter	403.0
Planetary Vehicle Weight Margin	713.3
Planetary Vehicle Plus Adapter Gross Weight	22,000.0

Table B-4

FLIGHT SPACECRAFT EQUIPMENT MODULE WEIGHT BREAKDOWN

Item	Weight (lb)
Equipment Module Structure	502.6
Capsule Support	18.8
Equipment Panels	100.0
Hinges	2.2
Latches	4.8
Mounting Rails	78.0
Structure Equipment Support	28.7
Meteoroid Protection Panels	208.8
Corner Members	8.6
Attachments and Miscellaneous	44.0
Miscellaneous Supports	8.7
Solar Array Support Linkage	-
Aft Equipment Module	-
Truss Members	-
Solar Array Supports	-
Thermal Control	132.2
Insulation	106.8
Louvers	17.1
Heaters and Thermostats	2.0
Attachments and Miscellaneous	6.3
Pyrotechnics	37.0
Release and Deployment System	7.7
Electrical Connectors	2.2
Explosive Valve Pyrotechnic (18)	0.6
Pyrotechnic Control Assembly (1)	25.0
Attachments and Miscellaneous	1.5
Power Supply	364.1
Solar Array	132.0
Battery (3)	138.0
Inverters	20.6
Battery Regulator (3)	42.0
Power Control Unit (1)	8.0
Shunt Element Assembly (2)	16.0
Power Distribution Box (1)	7.5

Table B-4 (Cont.)

Item	Weight (lb)
Integration	228.9
Cabling and Connectors (4)	190.0
Junction Box (1)	20.0
Umbilical	8.0
Cabling Channels	10.9
Guidance and Control	268.5
Gyro Reference Assembly (1)	10.0
Accelerometer (1)	1.0
Guidance and Control Electronics	13.0
Canopus Sensor (2)	12.0
Fine Sun Sensor (1)	0.2
Coarse Sun Sensor (4)	0.8
Earth Detector (1)	0.3
Solenoid Valve (16)	19.5
Pressure Vessel (2)	60.0
Nitrogen Gas	49.0
Regulator (4)	3.5
Thrusters (4)	4.0
Lines (2)	4.0
High-Gain Drive Assembly	32.0
Medium-Gain Drive Assembly	17.0
TVC Actuator (2)	36.0
Limb and Terminator Crossing Detector	1.2
Antenna Drive Electronics	5.0
Communications	125.5
Modulator Exciter (2)	6.0
Four-Port Hybrid Ring and Power Monitor (1)	0.6
One-Watt Transmitter and Power Monitor (1)	3.5
Power Amplifier Power Supply and RF Monitor (2)	15.6
Transmitter Selector (1)	1.0
S-Band Receiver (3)	15.0
Receiver Selector (1)	1.0
Circulator Switch (4)	7.3
Diplexer (3)	3.9
Low-Gain Antenna (1)	3.0
Medium-Gain Antenna (1)	13.4
High-Gain Antenna (1)	55.2
Telemetry and Command	90.5

Table B-4 (Cont.)

Item	Weight (lb)
Tape Recorders (6)	72.0
PCM Encoder (2)	8.0
Decoder (2)	5.3
Command Detector (2)	5.2
Computing and Sequencing	36.0
Balance Weights	15.0
Contingency (10 percent)	180.0
Gross Equipment Module Weight	1,980.3

Table B-5

FLIGHT SPACECRAFT PROPULSION MODULE WEIGHT BREAKDOWN

Item	Weight (lb)
Propulsion Module Structure	512.0
Lower Ring	11.6
Meteoroid Protection Panels	247.3
Reaction Control Supports	11.7
Attachments and Miscellaneous	14.3
Base Structure	25.0
Internal Structure	79.0
Corner Members	7.3
Tank Supports	84.0
Engine Supports	31.8
Trusses	-
Temperature Control	29.4
Insulation (Refrasil)	24.3
Heaters and Thermostats	2.0
Attachments and Miscellaneous	3.1

Table B-5 (Cont.)

Item	Weight (lb)
Engines and Valves	426.5
Injector	29.3
Combustion Chamber Assembly	280.0
Injector Pintle Actuator	4.0
Propellant Lines and Ducts	13.9
Electrical Harness	9.0
Instrumentation	2.7
Gimbal Assembly	27.2
Hardware-Engine Integration	9.4
Fuel Control Module	15.5
Oxidizer Control Module	18.0
Solenoid Valves (8)	14.0
Quad Check Valves (2)	1.0
Trim Orificies (2)	0.5
Filter (2)	2.0
Propellant Feed Assembly	363.1
Propellant Tanks (4)	292.3
Lines and Valves	48.2
Engine Start Tanks (2)	22.6
Pressurization System	414.4
Valves, Regulator, Etc.	33.2
Lines, Fill and Vent	13.2
Tank	368.0
Contingency	174.5
Residuals	462.5
Propellant (Including Start Tanks)	417.4
Helium	45.1
Propulsion Module at Burnout	2,382.4
Usable Propellant	11,071.0
Propulsion Module at Ignition	13,453.4

Table B-6

MOMENT OF INERTIA (AD 7-122)

Condition	Weight (lb)	Longitudinal CG, Z (in.)	Moment of Inertia (slug ft^2)		
			I_x	I_y	I_z
Without Capsule					
Ignition	16,597	114.4	7,322	5,679	9,525
Burnout	5,526	128.3	4,791	3,163	5,218
With Capsule					
Ignition	21,597	141.4	23,094	21,448	14,037
Burnout	10,526	177.1	15,265	13,626	9,729

B.5.3 Temperature Control

Temperature control of subsystems is maintained by a combination of spacecraft/sun orientation, insulation, surface coatings, and louvers. The earth-storable propellants are warmed by orientation to the sun to prevent freezing.

B.5.4 Internal Heat Sources

Various items of electronic equipment mounted in the equipment module represent a source of internally generated heat in the Voyager spacecraft. The thermodynamic definition of this heat source is as follows:

- Equipment power range = 392 w near earth and 319 w near Mars
- Uniform power density is assumed on equipment mounting panels
- Equipment temperature held to 75°F ± 15°F through an active thermo control system

Refer to weight tables for an equipment list.

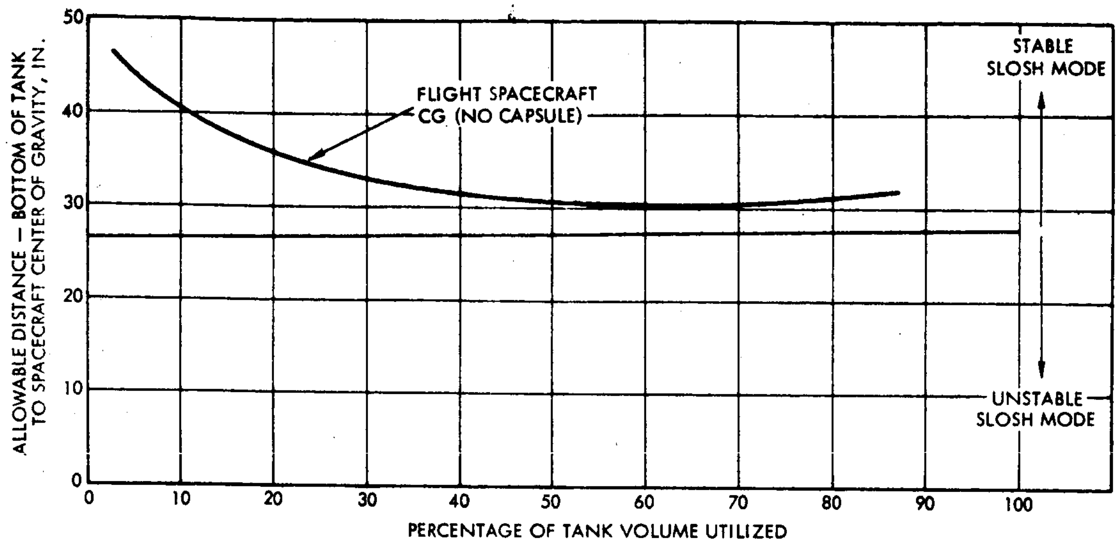


Fig. B-4 Allowable Center of Gravity Location

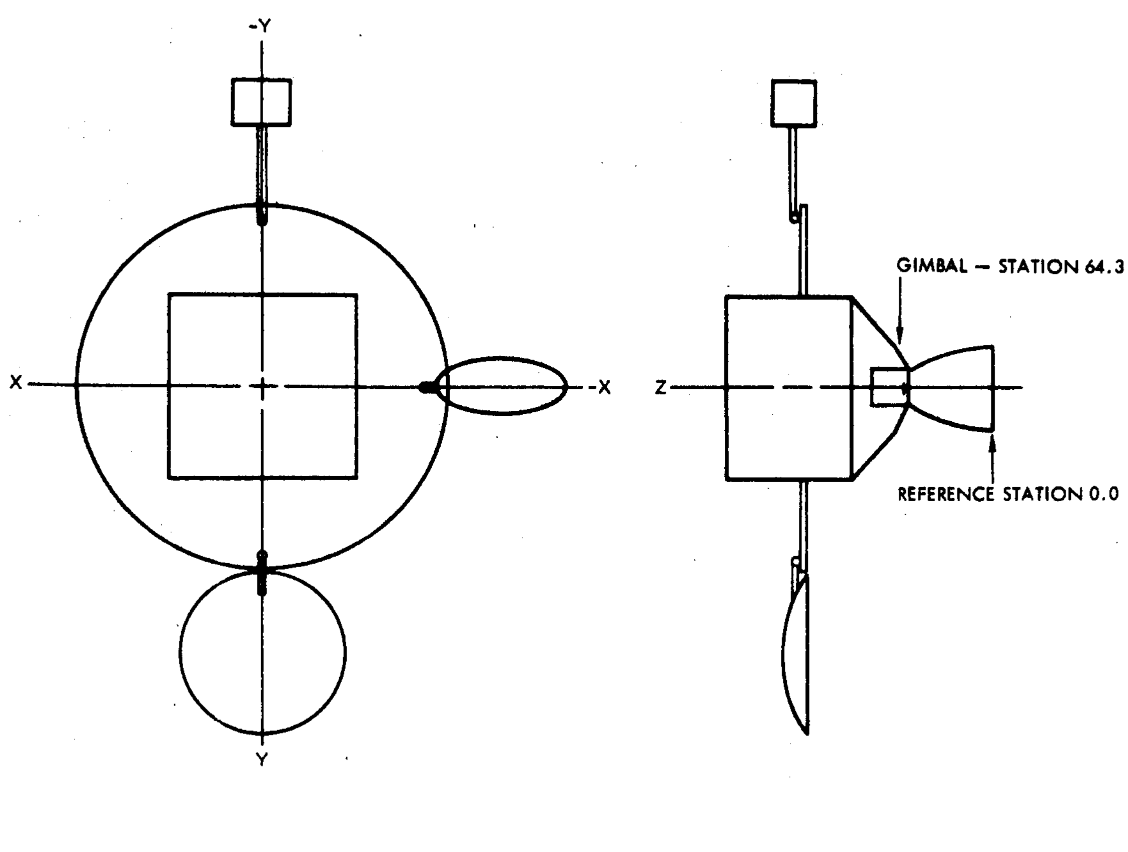


Fig. B-5 Axis Coordinate System

Appendix C
MARS EXCURSION MODULE BASELINE DESCRIPTION

C.1 INTRODUCTION

The Mars Excursion Module Ascent Stage, as defined by North American Rockwell in Ref. C-1, was chosen as one of the two reference spacecraft stages to be analyzed by Lockheed in Contract NASw-1644, Propellant Selection for Spacecraft Propulsion Systems. This document presents a summary description of the reference Mars Excursion Module (MEM) mission and spacecraft as extracted from Ref. C-1, and of the Aerobraker, which encloses the MEM while enroute from Earth to Mars, as extracted from Ref. C-2.

C.2 MISSION DESCRIPTION AND REQUIREMENTS

The MEM Ascent Stage must be capable of performing the following functions on a mission to Mars:

- Remain in a dormant state aboard the Mars Aerobraker from the time of launch from the surface, through 30 days in earth orbit, 160 days enroute to Mars while rotating at 4 rpm in the plane of the ecliptic, and aerodynamic entry of the Aerobraker into a 270 nm orbit about Mars. The Aerobraker configuration is shown in Fig. C-1, the mission profile is shown in Fig. C-2, and the structural temperature is shown in Fig. C-3.
- Accompany the MEM descent stage in a deorbit, aerodynamic entry, and propulsive landing on the surface of Mars.
- Remain in a standby condition on the surface of Mars for a period of 30 days while exposed to the atmosphere of Mars. The assumed model atmosphere is VM-7.
- Return the crew and crew module to a 270 nm rendezvous orbit with the Aerobraker. Figure C-4 shows the MEM mission profile from time of deorbit at Mars until return to orbit for rendezvous with the Aerobraker.

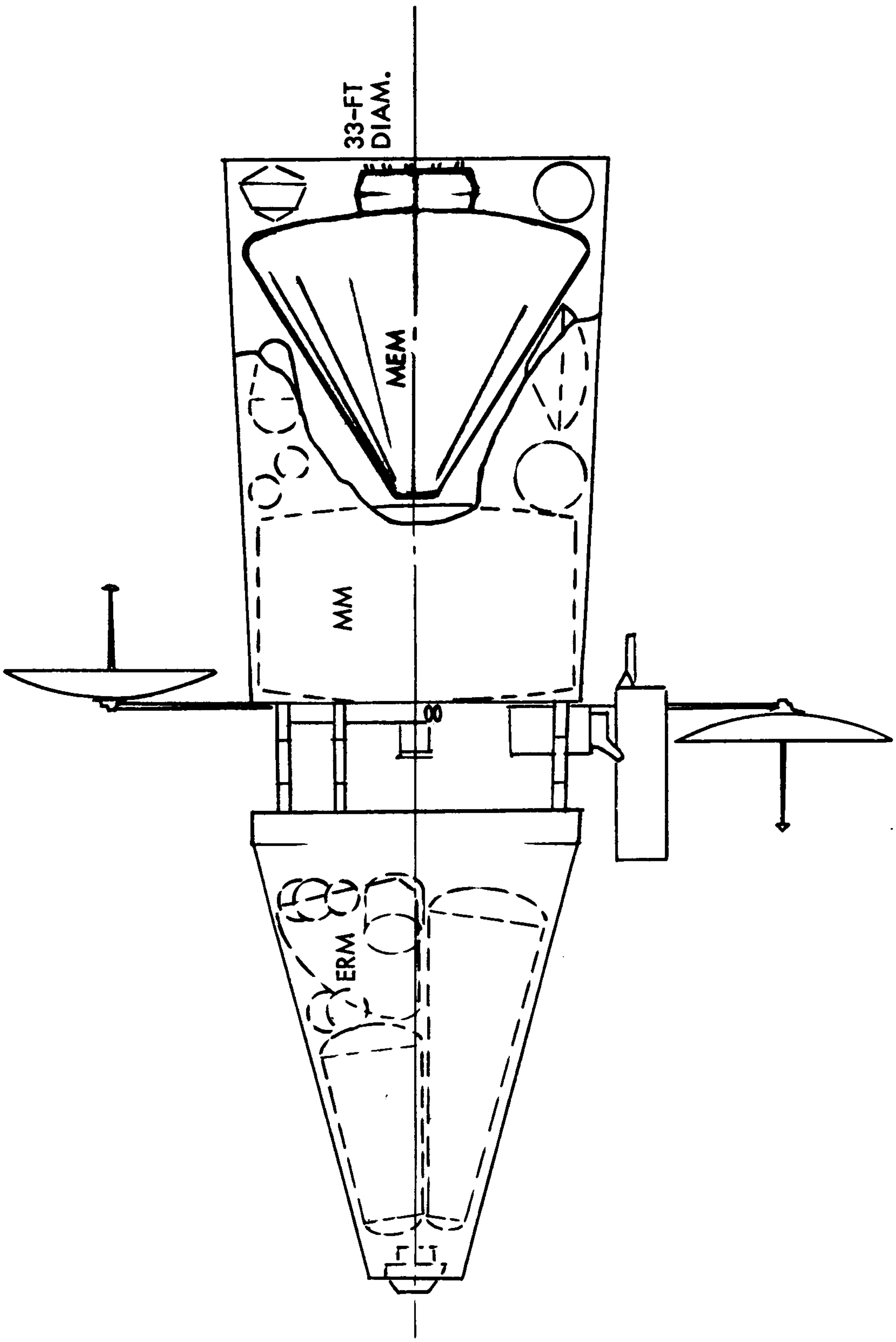


Fig. C-1 Baseline Aerobraker — North American Rockwell

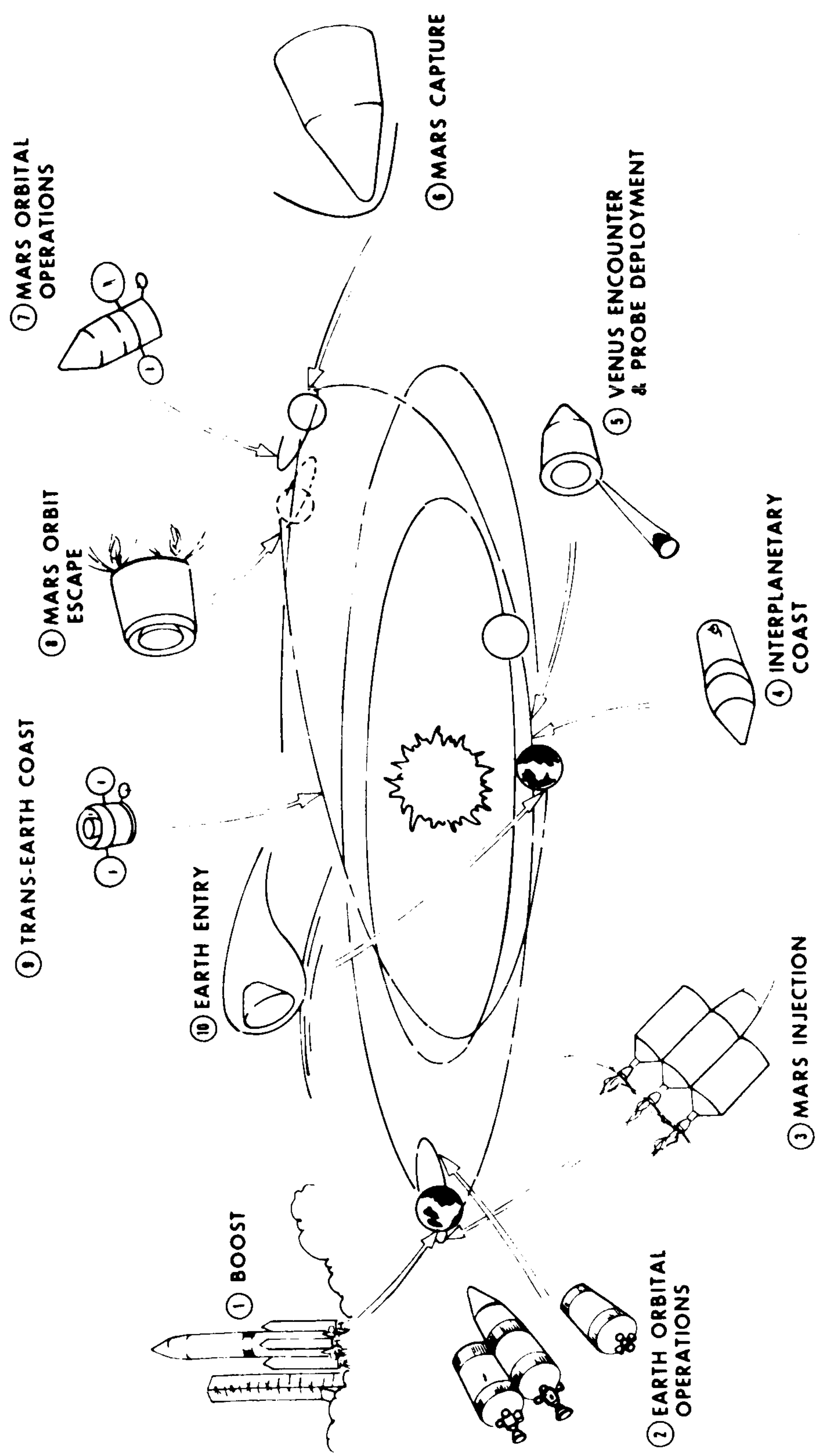


Fig. C-2 Mars Aerobraker Mission Profile

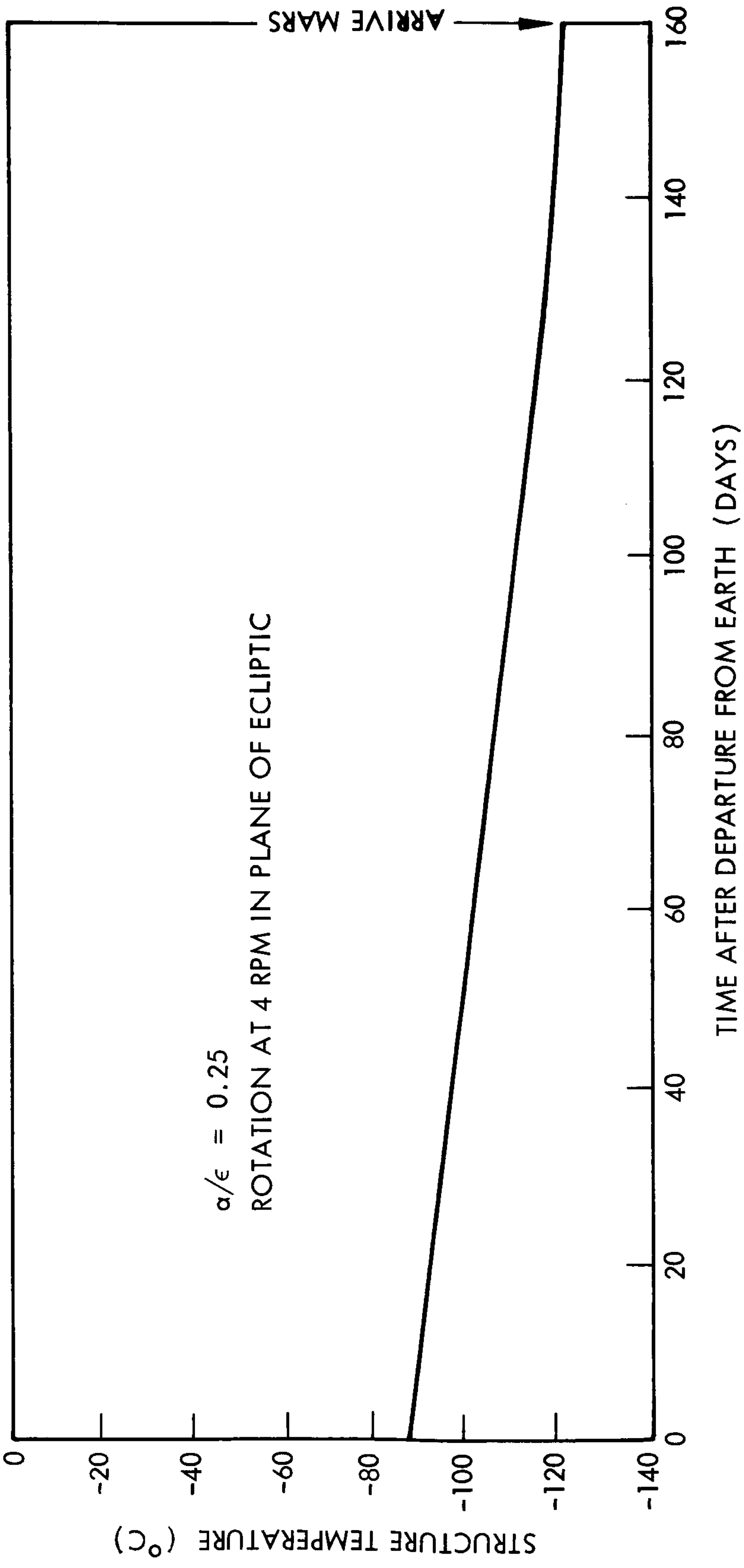


Fig. C-3 Mars Aerobraker Average Vehicle Structure Temperatures

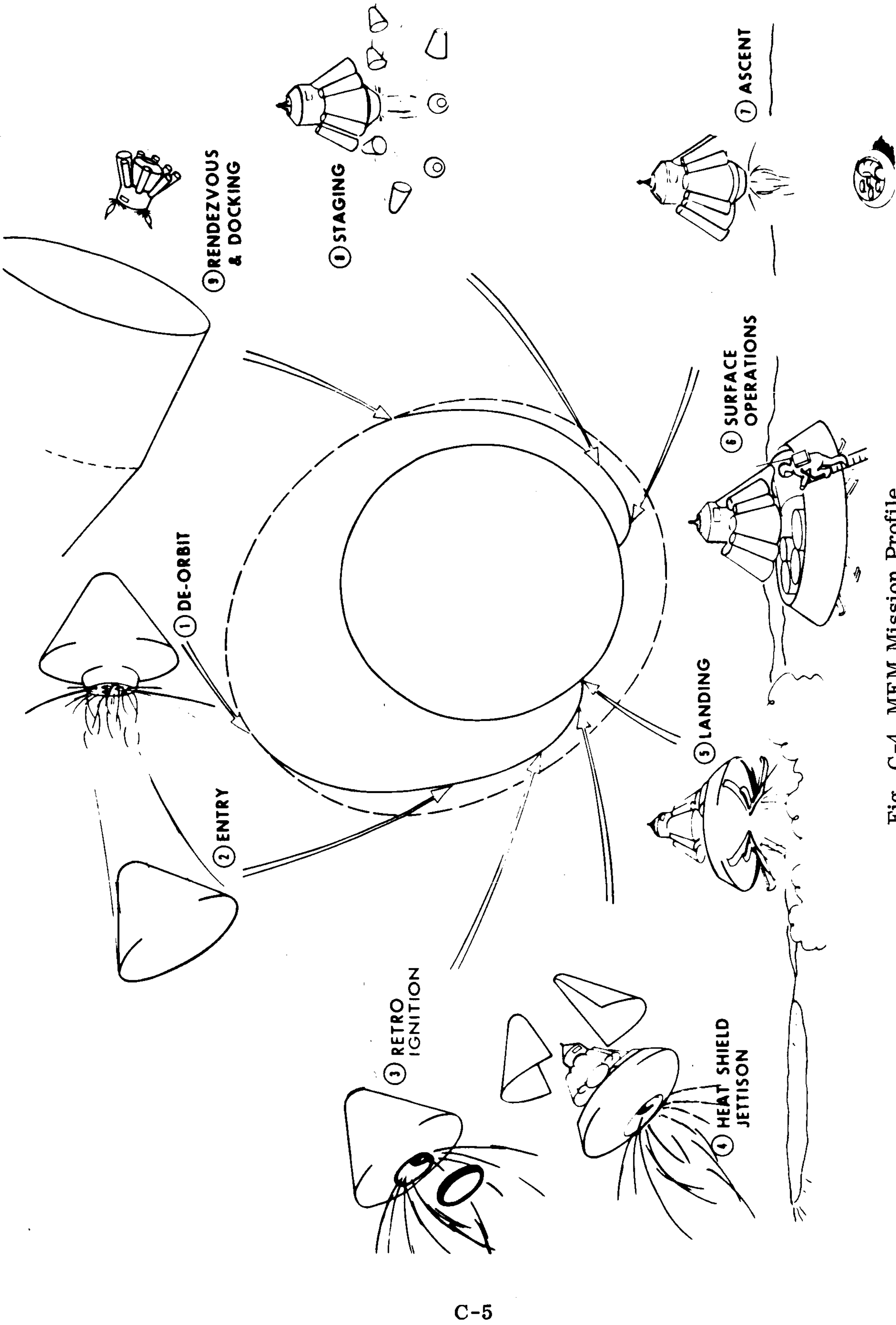


Fig. C-4 MEM Mission Profile

Maneuvers and velocity increments provided by the spacecraft propulsion system are given in Table C-1.

Table C-1

MEM ASCENT STAGE REQUIREMENTS

Maneuver	ΔV (Ideal) (ft/sec)
Ascent to 300,000 ft	13,800
Circularize at 100 nm	75
Transfer to 270 nm Circular Orbit	550
Contingency	1,443
Total	15,868

Maximum accelerations experienced by the spacecraft during launch and maneuvers are as follows:

Maneuver	Max Acceleration (Earth g's)
Earth launch, Mars entry, and landing	+5 axial, ±2 lateral
Mars Capture	-10 axial, ±3 lateral

C.3 DESIGN CONSIDERATION/CRITERIA

C.3.1 General

The spacecraft was configured based on the following general design considerations:

- The shape of the MEM should be that of the Apollo for aerodynamic entry at Mars.
- The maximum diameter of the MEM is 31.5 ft to fit inside the 33-ft Mars Aerobraker.
- The MEM should transport four men from Mars orbit to the surface of Mars, provide support for a 30-day stay, and then transport the men back to rendezvous with the Aerobraker in Mars orbit.

- The MEM descent stage and manned laboratory will be abandoned on the surface of Mars.
- The MEM ascent stage will incorporate a single engine of 30,000-lb maximum thrust and will use droppable first-stage propellant tanks.
- The optimum staging is achieved using a velocity of 9,000 ft/sec provided by the first stage tanks and 7,000 ft/sec by the second stage tanks.

C.3.2 Propulsion System

The ascent propulsion system design was based on use of a plug nozzle engine using FLOX/CH_4 propellant. Eight conical tanks with elliptical domed ends were used for the first stage and two ellipsoidal tanks for the second stage. Propulsion system parameters are described in Table C-2.

Table C-2
PROPULSION SYSTEM CRITERIA

Area	Item	Criteria
Propellant	Oxidizer	82.5% F_2, 17.5% O_2
	Fuel	Methane
	Mixture Ratio	5.75:1
	I_{sp}	383 sec
Engine	Expansion ratio	27:1
	Chamber Pressure	1,000 psia (pump-fed)
	Thrust	30,000 lb restartable
	Helium gas pressurization	20 to 30 psia
	Cooling	Combination of transpiration and ablative

Table C-2 (Cont.)

Area	Item	Criteria
Propellant Tanks and Insulation	Insulation density	4 lb/ft^3
	Insulation Conductivity	15^{-4} and 10^{-5} Btu/hr-ft-°F
	Insulation Evacuated on Surface of Mars	—
	Maximum Vapor Pressure	300 psi
	Heat Leaks Through Structure	30 percent of total
	Ullage Volume	6 percent
	Propellant Boiloff	None

C.3.3 Pressurization System

No data are available on the pressurization system.

C.4 CONFIGURATION DESCRIPTION

C.4.1 General Arrangement

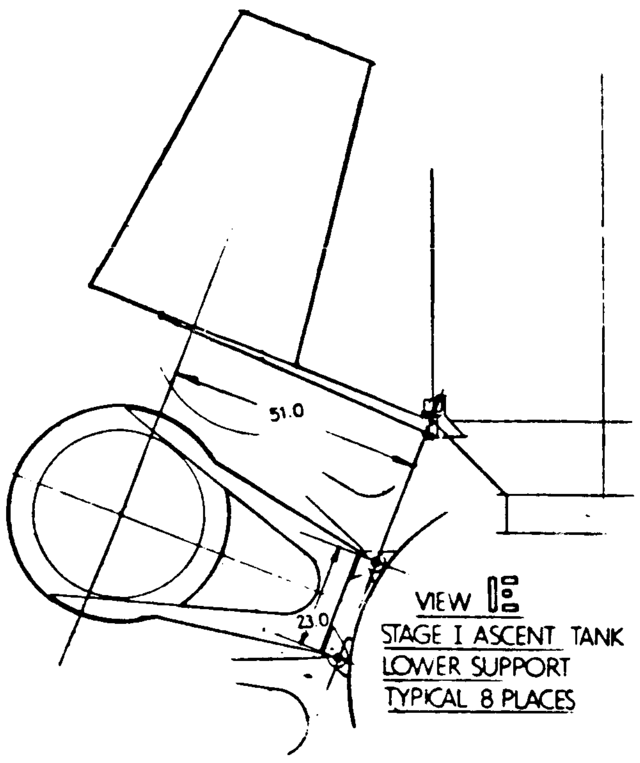
The general spacecraft arrangement is shown in Fig. C-5. The MEM is 30 ft in diameter. The total weight of the MEM Ascent Stage is 24,600 lb, including ascent capsule and ascent propulsion stages I and II. The ascent stage separates from the descent stage and laboratory at the start of ascent, as shown in Fig. C-5. Note that weights shown on Fig. C-5 are for an ascent ΔV of 20,350 ft/sec, rather than the nominal 16,000 ft/sec.

C.4.2 Weight Breakdown

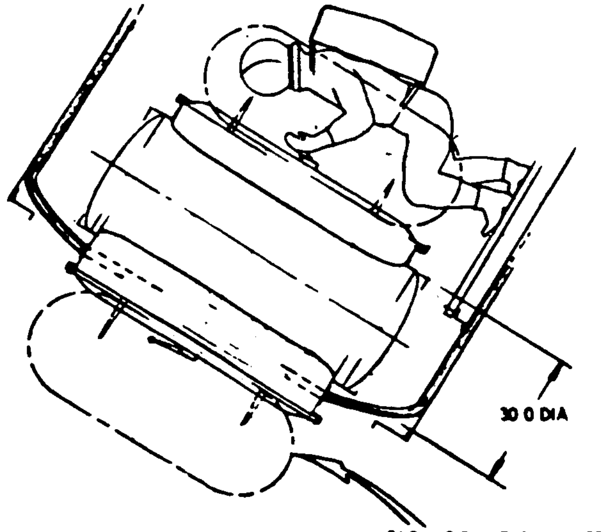
A weight summary for the ascent stage is given in Table C-3.

FOLDOUT FRAME 1

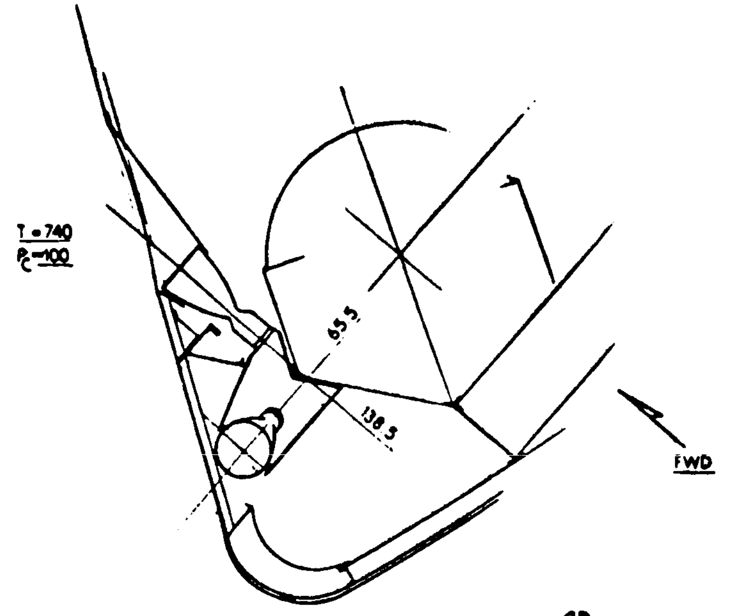
FOLDOUT FRAME 2



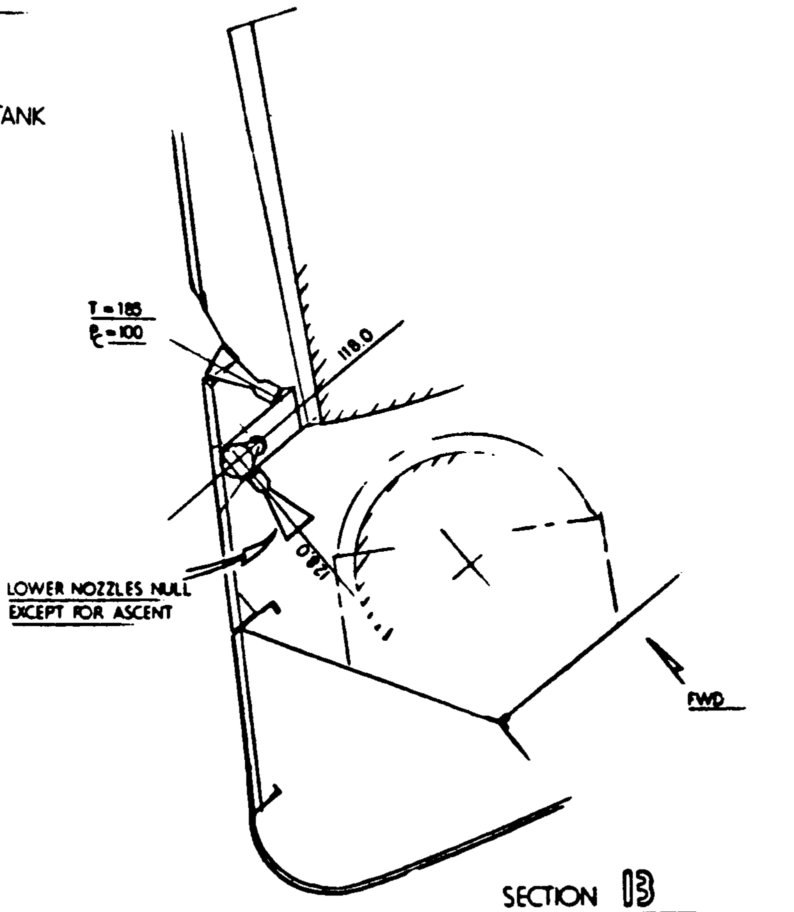
VIEW E
STAGE I ASCENT TANK
LOWER SUPPORT
TYPICAL 8 PLACES



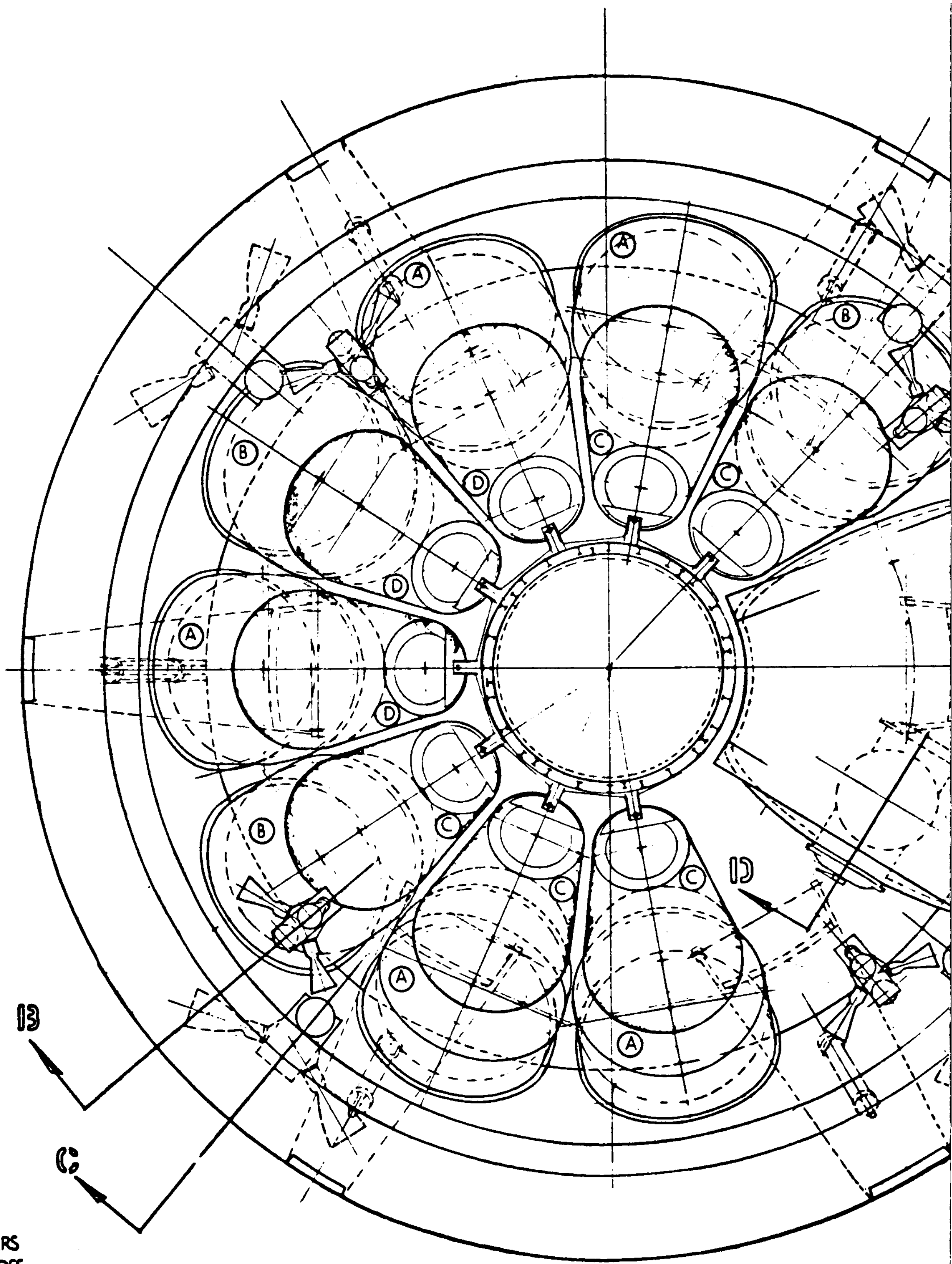
SECTION D
LIVING MODULE AIRLOCK



SECTION C
TYPICAL OF FOUR RCS CLUSTERS
FOR ENTRY ATTITUDE



SECTION B
TYPICAL OF FOUR RCS CLUSTERS
FOR ORBITAL & ASCENT ATTITUDES



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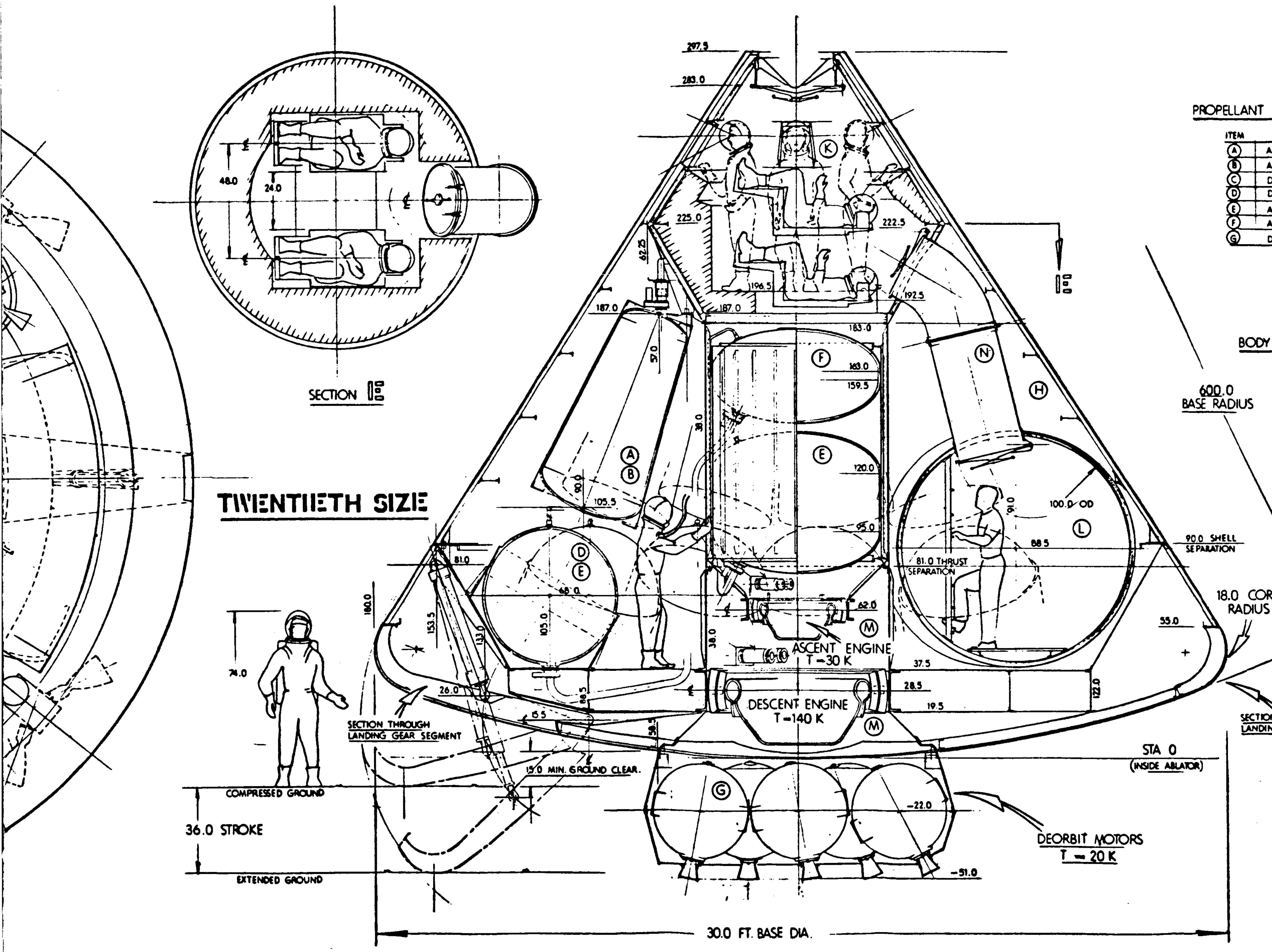

PROPELLANT TANKS & TANK NET VOLUMES

ITEM	STORAGE	NO. TANKS	VOL. EACH
A	ASCENT STAGE I OXIDIZER	5	51.0 FT³
B	ASCENT STAGE I FUEL	3	51.0
C	DESCENT OXIDIZER	5	53.0
D	DESCENT FUEL	3	53.0
E	ASCENT STAGE II OXIDIZER	1	98.0
F	ASCENT STAGE II FUEL	1	52.0
G	DEORBIT STAGE (SOLID)	7	16.6

BODY VOLUMES

ITEM	DESCRIPTION	VOLUME
H	BODY INSIDE SKIN	7200. FT³
K	FLIGHT STATION	342.
L	LIVING MODULE	460.
M	ENGINE SECTION	196.
N	TUNNEL	25

PROPELLANTS

DE ORBIT	6 600 LB	SOLID
DESCENT	23,200 LB.	FLOX/METHANE
ASCENT STAGE I	22,300 LB.	FLOX/METHANE
ASCENT STAGE II	8,600 LB.	FLOX/METHANE

$W_O/W_F = 5.75$

ULLAGE – 6% MIN.

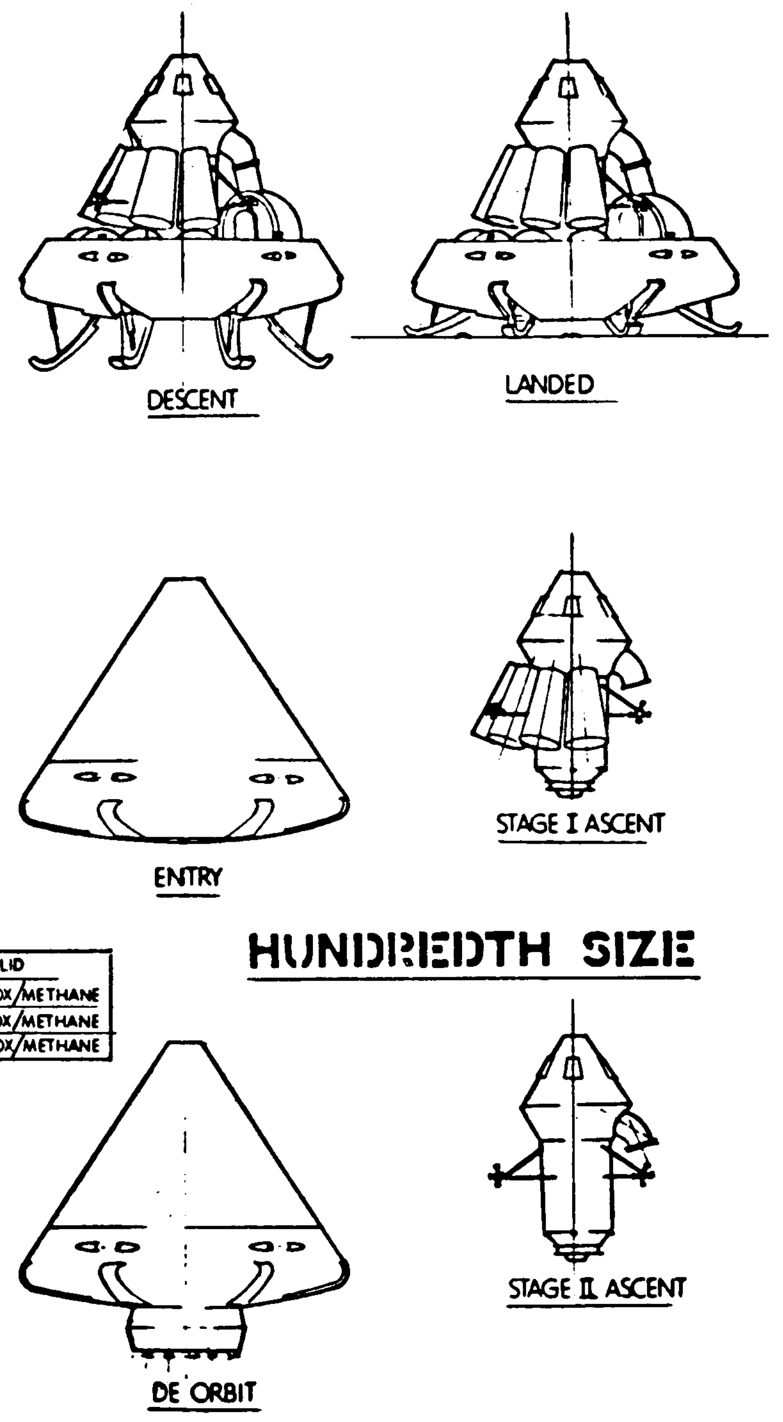

Fig. C-5 Mars Excursion Module General Configuration

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Table C-3

ASCENT STAGE WEIGHT SUMMARY

Item	Weight (lb)
Ascent Capsule[a]	(5,260)
Primary Structure	560
Couch, Restraints	80
Hatches, Windows	120
Docking Provisions	170
Panels, Supports	50
Battery (10 kw-hr)	270
EPS Distribution	230
Communication	210
Guidance and Navigation	225
Controls and Displays	200
Instrumentation	190
Life Support System	950
RCS (Dry)	290
RCS (Propellant)	240
Return Payload	300
Crew (90 Percentile)	700
Contingency	475
Stage II Ascent	(6,510)
Tanks and System	400
Engine Installation	300
Contingency	70
Propellant	5,740
Stage I Ascent	
Tanks and System	(12,830)
Contingency	830
Propellant	80
Total Ascent Stage	24,600

(a) At ascent liftoff, the CG is 147 in. from the forward face of the descent stage heat shield and on the center line.

C.5 SUBSYSTEMS DESCRIPTION

A description of the spacecraft subsystems is limited here to those items influencing the propellant selection study configurations through physical arrangement, thermal effects, or structural protection.

C.5.1 Meteoroid Protection

Meteoroid protection is not specified.

C.5.2 Temperature Control

Temperature control of the propulsion subsystem is maintained by insulation on the propellant tanks.

C.5.3 Internal Heat Sources

Internal heat source are not identified.

C.6 REFERENCES

C-1 North American Rockwell Corp., "Definition of Experimental Tests for a Manned Mars Excursion Module," Phase I and II and Final Reports, Contract NAS9-6464

C-2 -----, Space Division, Draft Final Report SD 67-994-2, Vol. II: Technical Analysis, "Study of Technology Requirements for Atmosphere Braking to Orbit About Mars and Venus," Jan 1968